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CATAPULT AND ARRESTED LANDING FATIGUE TESTS OF THE MODEL E-2A/B AIRPLANE

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The right-hand keel beam failed after 4556 catapult test cycles. Using a test scatter factor of 2, this is equivalent to 2278 service catapult launches. The keel beam was repaired and modified and testing was continued until a total of 8000 catapult test cycles had been applied.

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S U M M A R Y

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INTRODUCTION

The E-2 airplane was certified for 1000 catapult launches and 1000 arrested landings as a result of catapult and arrested landing fatigue tests performed by the GAC (Grumman Aerospace Corporation) in 1963 and 1964. The results of these tests are presented in references (a) and (b). This test program was initiated because a review of service usage records indicated that certification for 1000 arrested landings and catapult launches would be grossly insufficient to satisfy projected operational requirements for this model airplane. These projected requirements showed a need for the capability to sustain the effects of 3000 arrested landings and catapult launches. Therefore, the objectives of the test program at its inception were to determine the capacity of the E-2 airframe to safely sustain the effects of 3000 arrested landings and catapult launches and to determine any structural modifications necessary to achieve this life. As the program was nearing completion, however, the projected operational requirements for this model airplane were revised, indicating a need for 4000 arrested landings and catapult launches rather than 3000. This revision came too late to effect a change in the arrested landing test, however, the objectives of the catapult test program were changed to satisfy this new requirement. It should be noted, however, that the calculated arresting fatigue life, reported in reference (c), is in excess of 4000 arrested landings.

DESCRIPTION OF TEST SPECIMEN

The test specimen was an E-2 fuselage and wing center section that had been used by GAC in their arrested landing and catapult static tests and in drop tests. During one drop test a major fuselage failure occurred between FS (Fuselage Station) 207 and FS 275. This failure was repaired by GAC and additional drop testing was performed with this fuselage, however, the repair is not a duplicate of the original airplane structure. The test specimen is shown in figure 1.

The E-2 is a nose tow catapult type airplane. The catapult forces on the nose gear tow link are transmitted by the shock strut and drag brace to support fittings attached to the fuselage. The shock strut support fittings, located at FS 64.50, provide vertical and lateral stability while fore and aft stability is provided by the drag brace support fittings, located at FS 126.50.

A dummy nose gear was used in this test instead of an actual nose gear since the catapult fatigue life of the nose gear was established in the GAC tests reported on in reference (a). The dummy nose gear provided the correct distribution of loads to the shock strut support fittings and drag brace support fittings.

Arresting loads are introduced into the aft fuselage through a tubular A-frame arresting hook. These loads are reacted by fittings attached to

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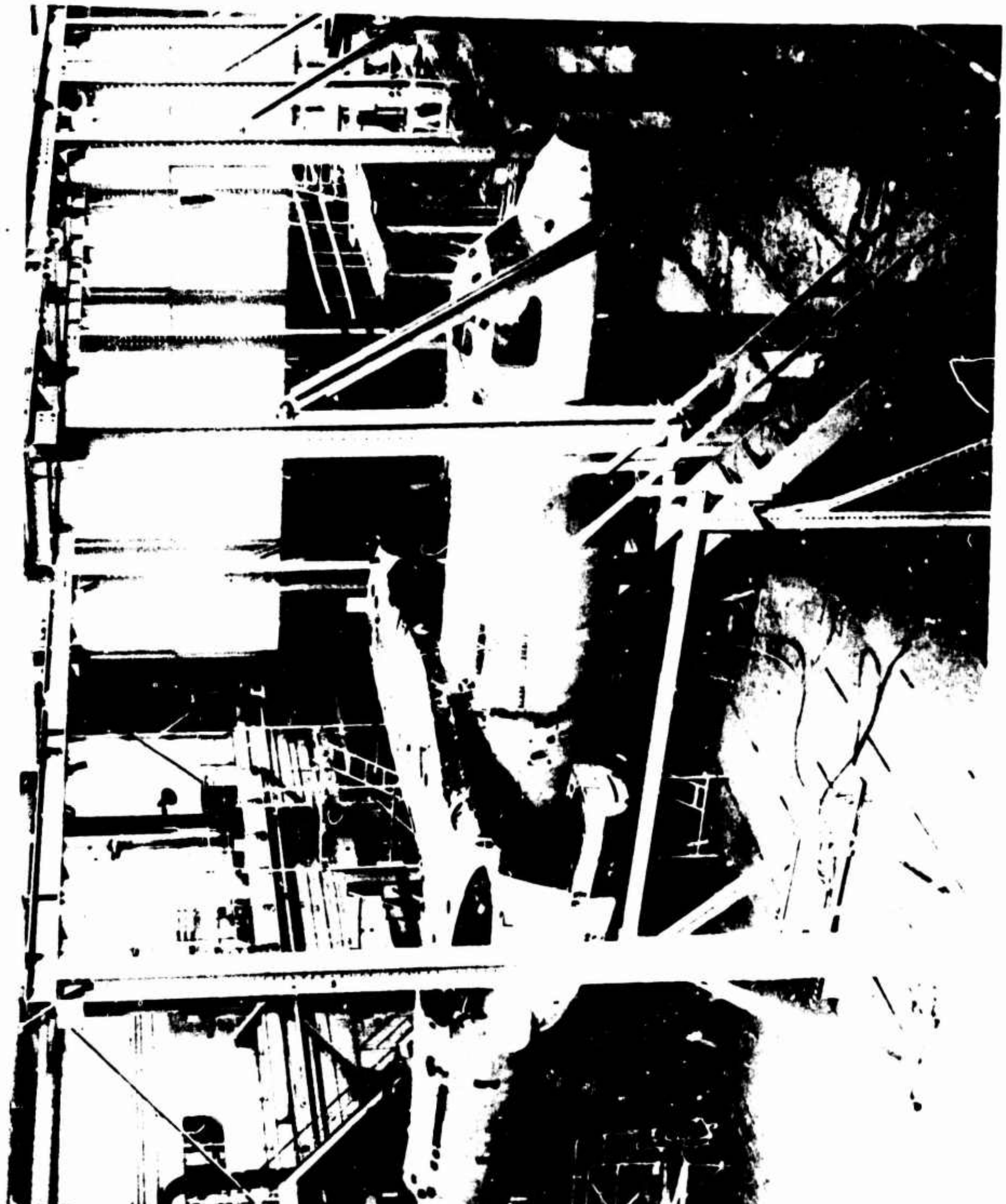

FIGURE 1 - Test Specimen

the fuselage bulkhead at FS 599.50. Vertical and side loads acting on the fittings are reacted by the bulkhead while longitudinal loads are transmitted to the lower longerons. A new A-frame arresting hook was used in this test.

TEST PROGRAM

The test program consisted of two parts, an arrested landing fatigue test and a catapult fatigue test, performed seperately in that order.

The details of each test are as follows:

ARRESTED LANDING TEST

To demonstrate the capability to withstand the effects of 3000 arrested landings in service, using a scatter factor of 2, a total of 6000 arrested landings were simulated during the test.

The locations and magnitudes of applied loads and reactions are shown in appendix A, figure A-2 and table A-1. The loads correspond to GAC condition 12F arrested landing, free flight engagement with maximum hook load and mean run-out for an arrested landing design gross weight of 40,660 pounds. Loads were applied in accordance with the test spectrum shown in table I. Each cycle in this spectrum represents one arrested landing. Each block consists of 100 cycles.

TABLE I

ARRESTED LANDING TEST SPECTRUM

Load Block	No. Of Cycles	A	S_R	S_L
	40	68.3	37.7	
	5	68.3	75.4	
1	45	90.7	37.7	
	5	90.7	75.4	
	4	117.8	37.4	
	1	117.8	75.4	
	40	68.3		37.7
	5	68.3		75.4
2	45	90.7		37.7
	5	90.7		75.4
	4	117.8		37.7
	1	117.8		75.4

A = Percent limit aft load
S_R = Percent limit side load applied to the right
S_L = Percent limit side load applied to the left

Each cycle was applied as shown in figure 2.

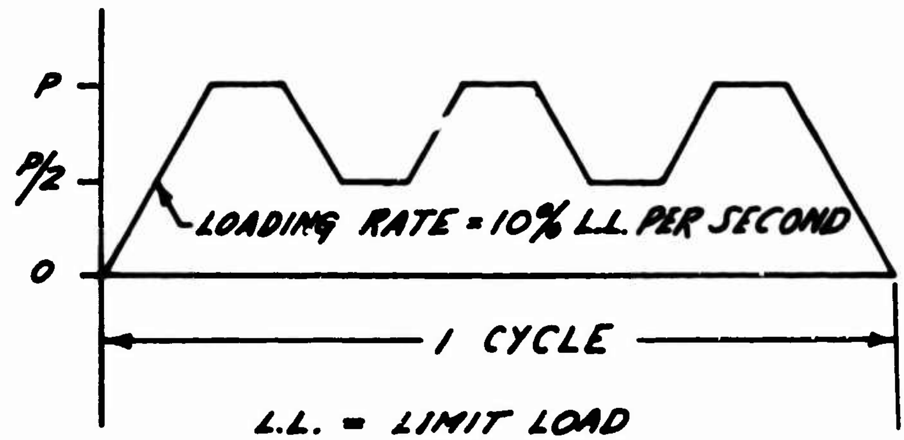

FIGURE 2 - Typical Arrested Landing Test Load Cycle

CATAPULT TEST

To demonstrate the capability to withstand the effects of 4000 catapult launches in service, a total of 8000 catapult launches were simulated during the test.

The locations and magnitudes of applied loads and reactions are shown in appendix A, figure A-9 and table A-4. The loads correspond to GAC condition 11c, Catapult Start of Run II, for a catapulting design gross weight of 47,940 pounds.

Each test load cycle, representing 1 catapult launch, was applied according to the requirements of paragraph 3.5.3.1 of MIL-A-8867. A graphical representation of the test load cycle is shown in figure 3.

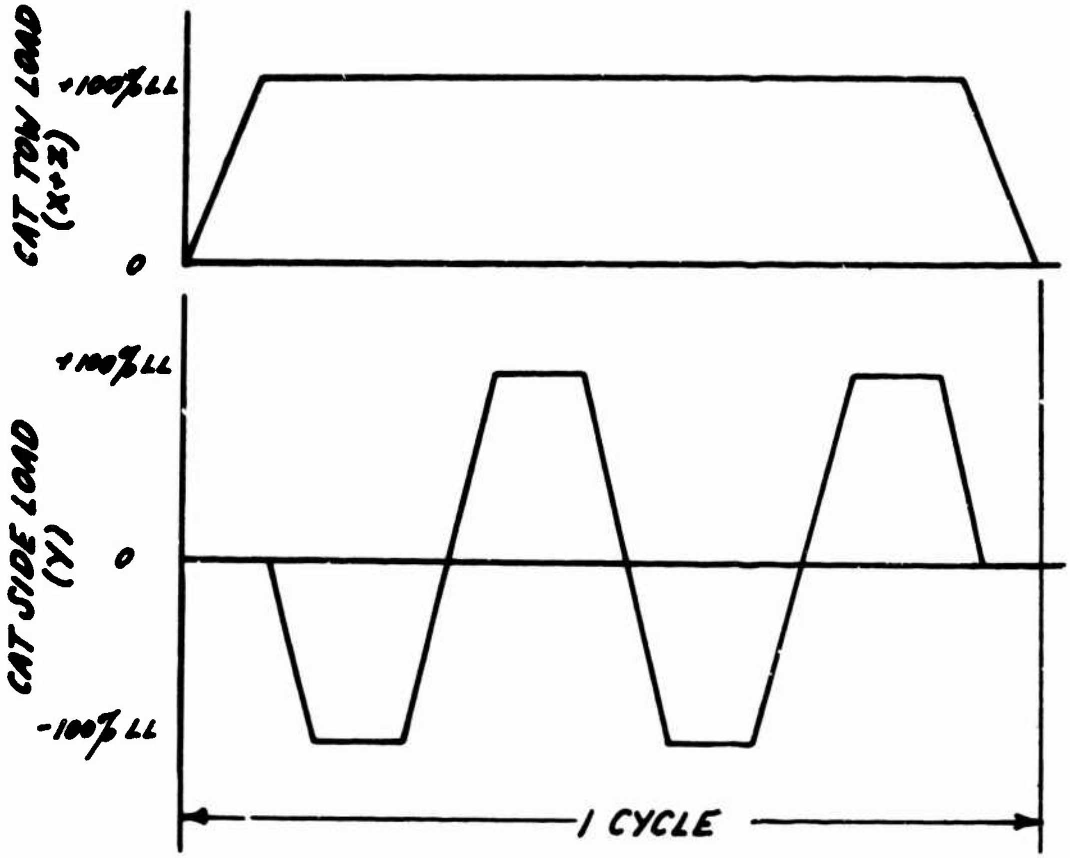

FIGURE 3 - Typical Catapult Test Load Cycle

TEST METHOD

The test specimen was supported by dummy main landing gear which reacted loads in the vertical, axial and lateral directions. A third reaction point which was designed to react vertical loads only was provided at either the nose gear shock strut support fitting (arrested landing test) or the frame at FS 555 (catapult test). The specimen was positioned so that the FRL (Fuselage Reference Line) was parallel to and 140.9 inches above the test floor and the plane of symmetry was perpendicular to the floor.

Test loads were applied to the specimen with hydraulic actuators which were part of an electro-hydraulic, servo-controlled closed loop loading system. Independent control of each actuator was provided by individual servo-valves and servo-controllers. Load direction and phase relationships for the actuators were provided by a multichannel programmer.

Loads were monitored on chart recorders and a multichannel bar graph video display, all of which provided overload protection. Additional and independent overload protection was provided by error detectors on each servo-controller and stroke limit switches on each actuator. Triggering any overload system would immediately dump hydraulic pressure at each actuator and at the hydraulic power supply.

Additional test method details, particular to each test condition, are shown below.

ARRESTED LANDING TEST

The arresting hook loads were applied to the apex of the A-frame through a special fitting that replaced the stinger. The X and Z components of the arresting load were applied as a resultant using a single actuator. The sideload was applied to the same fitting using a separate actuator. The overall arrested landing test set-up is shown in figure 4.

CATAPULT TEST

The X and Z components of the catapult load were applied to the dummy nose gear as a resultant using a single actuator. The sideloads were applied with a separate actuator. The catapult loading system is shown in figure 5. An overall view of the catapult test set-up is shown in figure 6.

RESULTS

ARRESTED LANDING TEST

A total of 6000 arrested landing cycles (60 blocks) were applied to the test specimen with no indications of structural failure.

CATAPULT TEST

During cycle number 2798, a skin failure occurred on the left side between FS 150 and FS 159. This failure, shown in figures 7 and 8, occurred because of extensive skin corrosion in that area. Figure 9, which is a close-up of one of the fracture surfaces, clearly shows the material degradation caused by the corrosion. Examination of skin revealed other corroded areas aft of the failure, therefore, a section of skin extending from FS 141 to FS 203 was cut out and a .050 7075-T6 repair patch was installed. This repair is shown in figure 10.

During cycle number 3910, a loud noise occurred and the test was stopped so that the cause of the noise could be located. Inspection revealed that the fuselage frame at FS 183 (stbd. side) had failed. The failure is shown in figure 11. The frame was repaired and testing

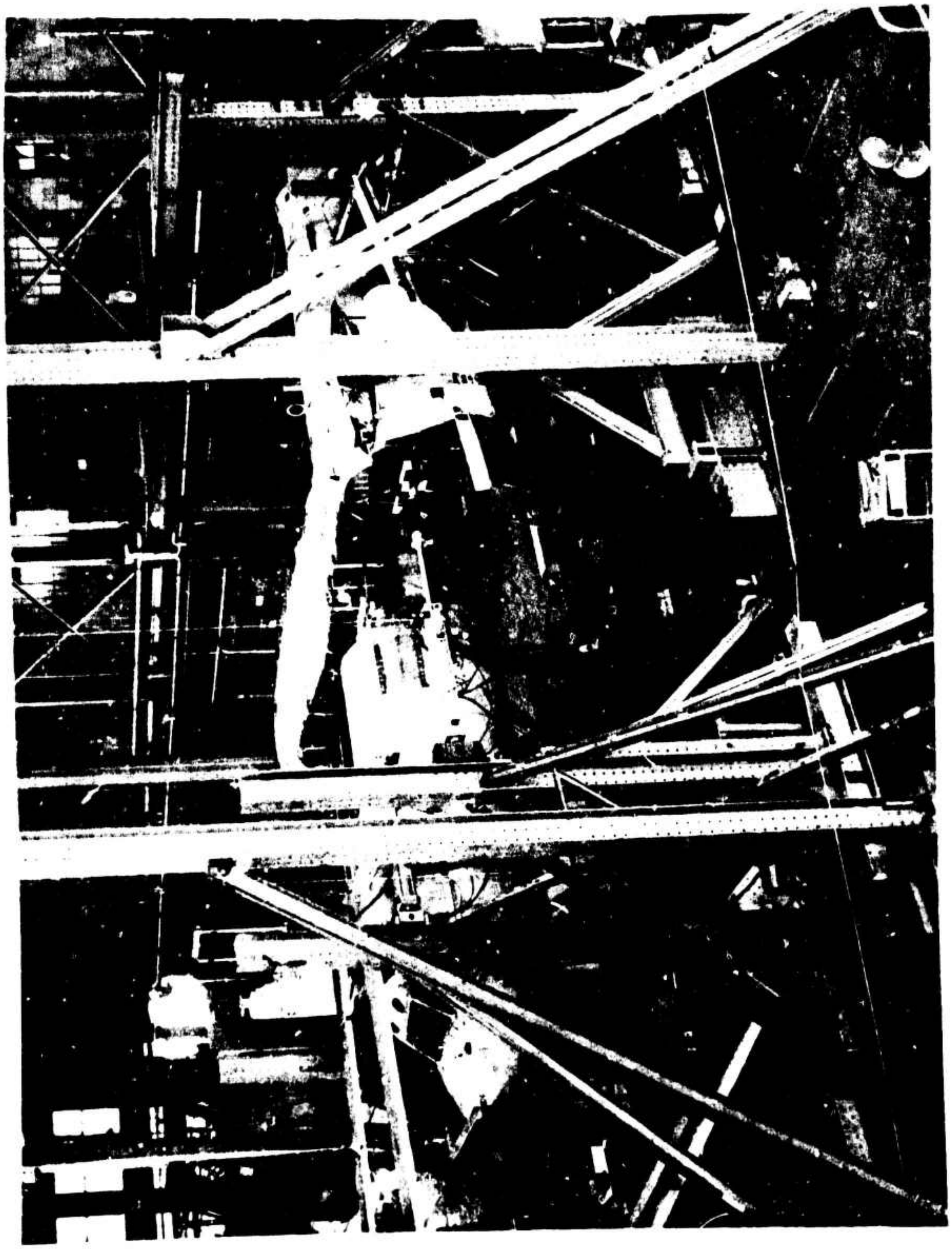

FIGURE 4 - Arrested Landing Test Set-Up

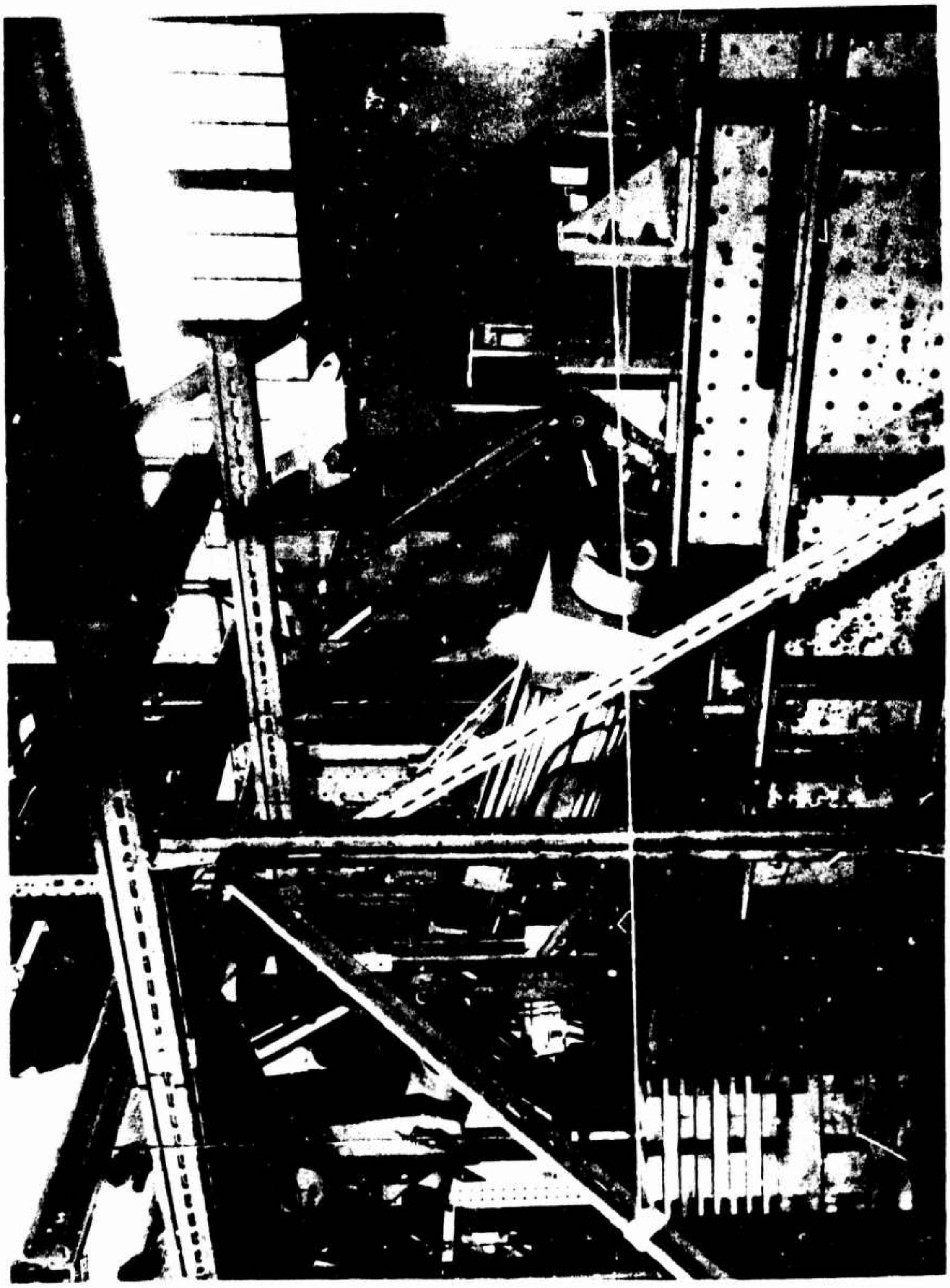

FIGURE 5 - Catapult Loading System

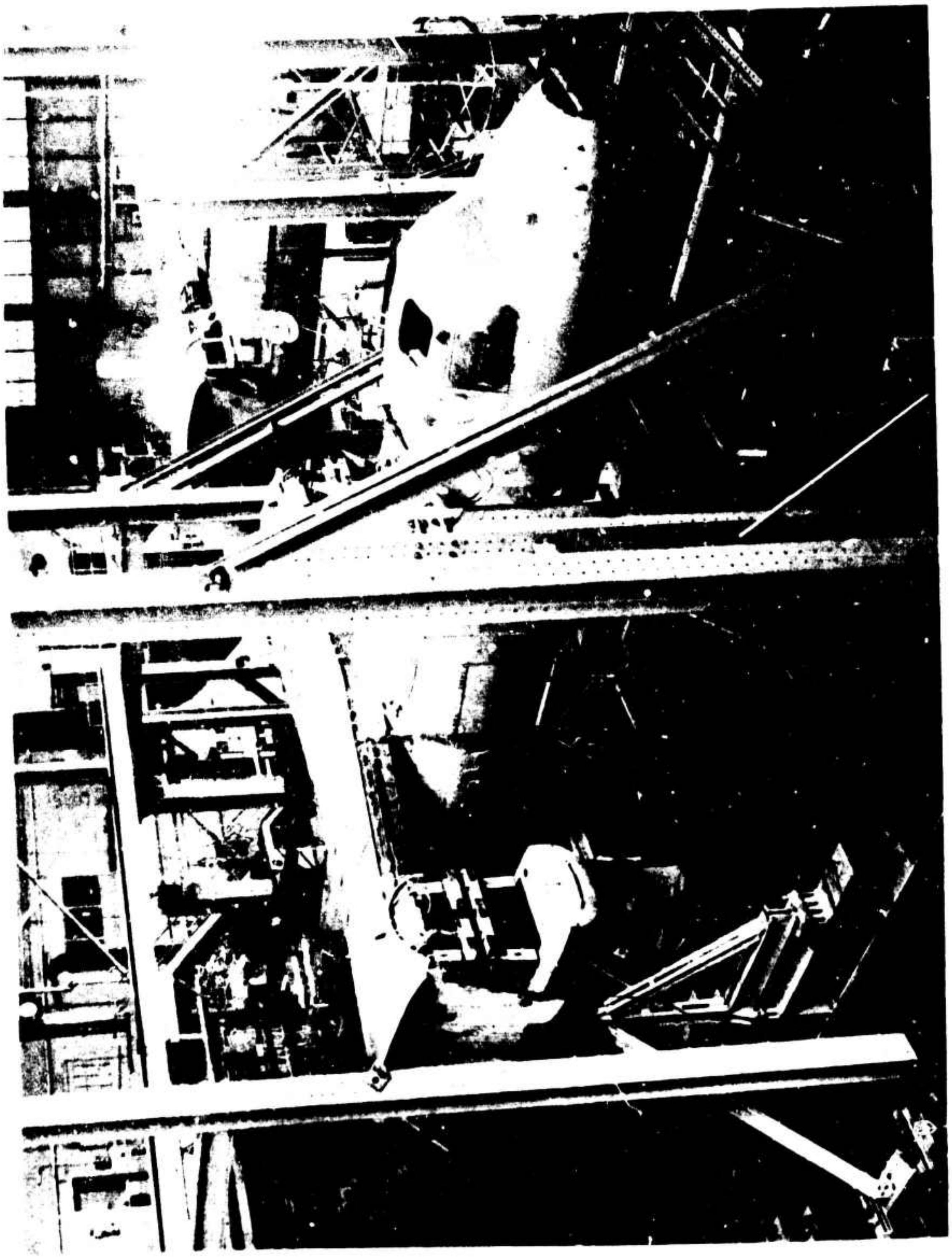

FIGURE 6 - Catapult Test Set-Up

FIGURE 7 - Skin Failure - Cycle No. 2798

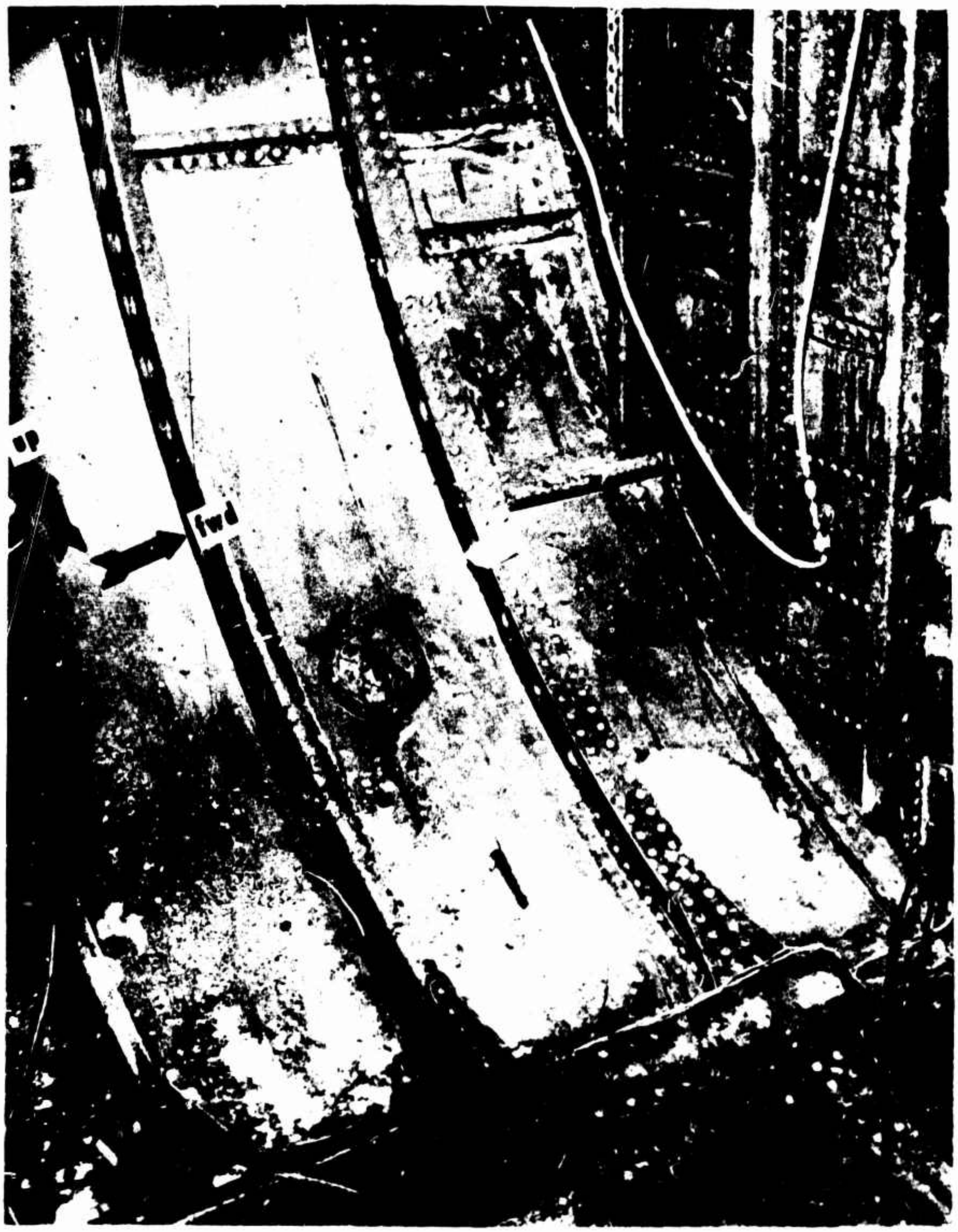

FIGURE 8 - Skin Failure - Cycle No. 2798 - View Inside Fuselage

FIGURE 9 - Close-Up of Skin Failure

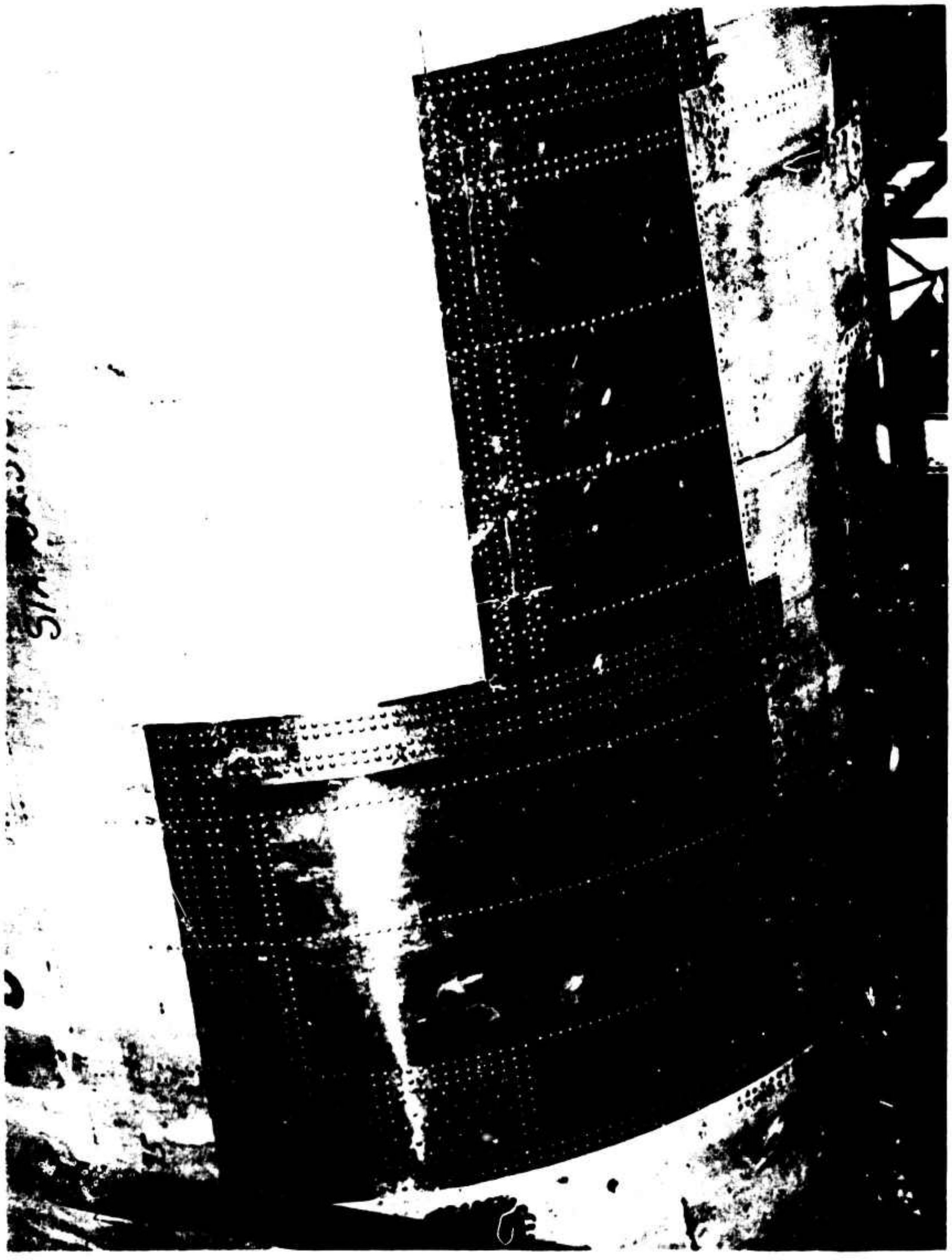

FIGURE 10 - Skin Repair

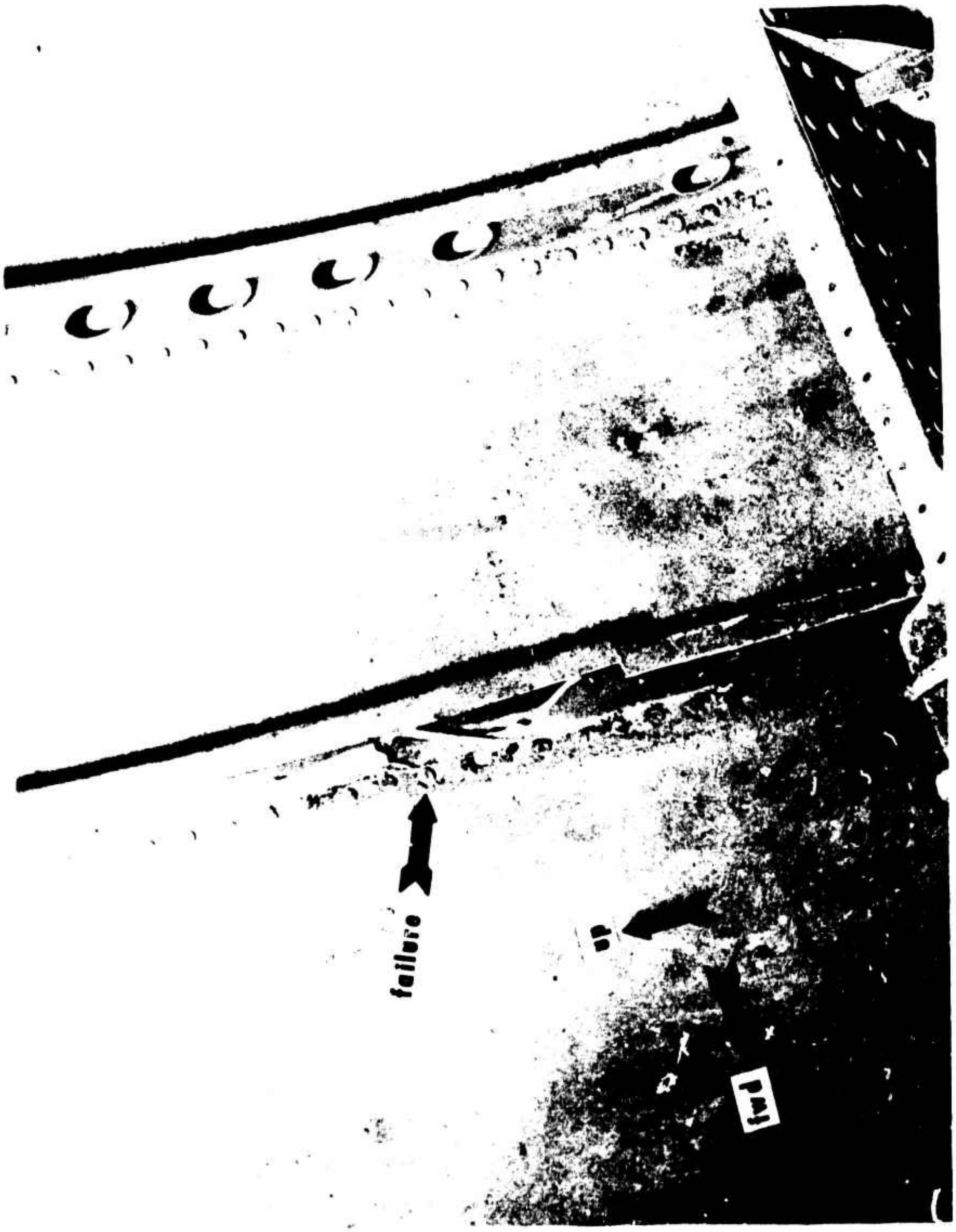

FIGURE 11 - Fuselage Frame FS 183 - Failure at Cycle 3910

resumed. Examination of the fracture surface indicated that the failure originated at a rivet hole (see figure 11) and propagated along the radius between the web and lower flange and finally through the web and upper flange of the frame. The origin of the crack was heavily corroded indicating that the crack had existed for quite some time and had originated prior to this test. This theory is further substantiated by the fact that there was a dent on the outside of the fuselage at the exact location of the origin of the failure before this test began.

During cycle number 4556, a major structural failure occurred. This failure originated in the right-hand keel beam at FS 182 (see figure 12) and continued into the adjacent skin resulting in the damage shown in figures 13, 14, and 15.

Visual inspection of the keel beam fracture surface revealed a dark area adjacent to a rivet hole in the web suggesting that this was the failure origin and that there had been a crack at that point for a long period of time. However, it was decided that a metallurgical examination would be necessary to determine the exact mode of failure.

Since 4556 cycles were far short of the desired 8000 cycles, a meeting of representatives of the Naval Air Systems Command, GAC, and the Naval Air Development Center was held to determine the future course of action for this program.

It was mutually agreed that the most feasible course to follow would be to repair the failure, install a modification on the right-hand keel beam and continue testing. Assuming that 8000 cycles were successfully completed, the keel beam modification would be installed on every E-2 in service at the appropriate time during its life. Details of the modification, a .063" 7075-T6 doubler, are shown in figure 16. The left-hand keel beam, which is of heavier construction, is shown for comparison.

The completed repair is shown in figures 17 and 18. The keel beam modification is shown in figure 19.

Prior to the resumption of testing, an in-depth inspection was performed, including radiographic examination of the left-hand keel beam. There were no indications of any structural damage.

Testing was resumed and continued to completion (8000 total cycles) with no additional structural failures.

The metallurgical examination of the keel beam fracture surfaces found that the failure originated at the central rivet hole in the web (see figure 16). Cause of the failure was attributed to a pre-existent crack, i.e., a crack that existed in the keel beam prior to the start of this program. This crack developed by a fatigue mechanism, most likely during the repeated GAC drop tests. During the several years between the end of the GAC drop tests and the beginning of this program, the

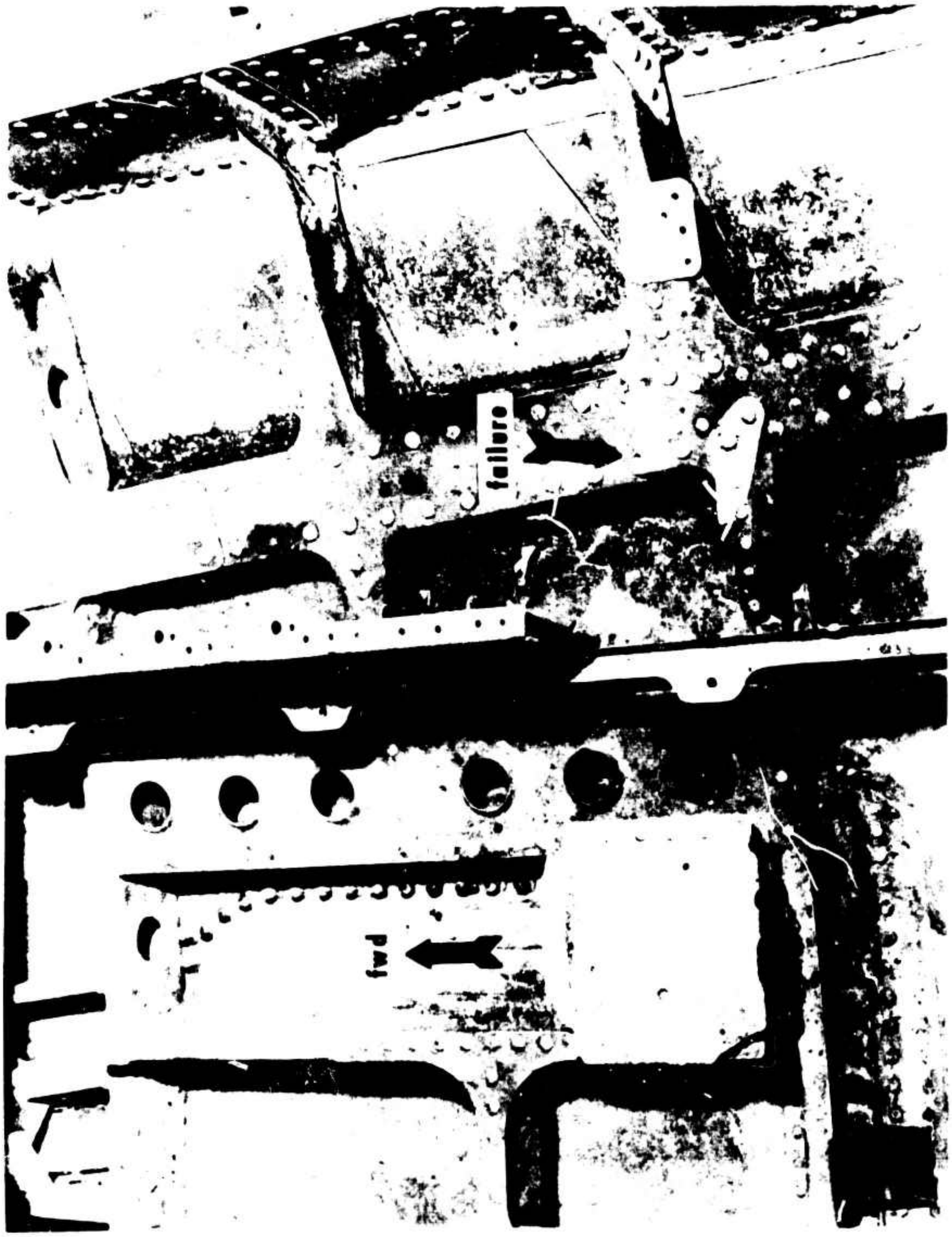

FIGURE 12 - R. H. Keel Beam Failure - Cycle No. 4556

FIGURE 13 - Failure at Cycle No. 4556 - Looking Inboard at Skin

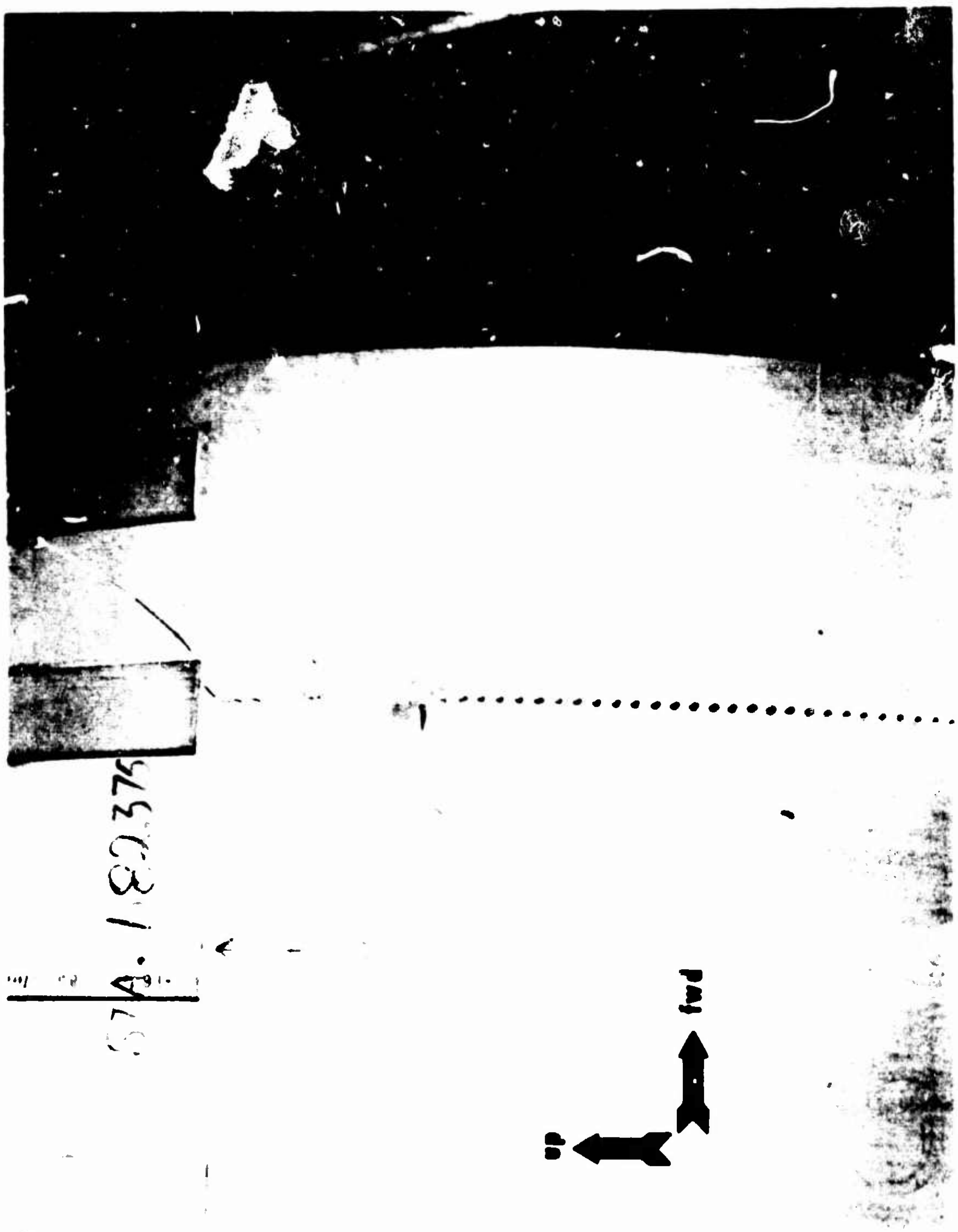

FIGURE 14 - Failure at Cycle No. 4556 - Looking Inboard at Upper Skin

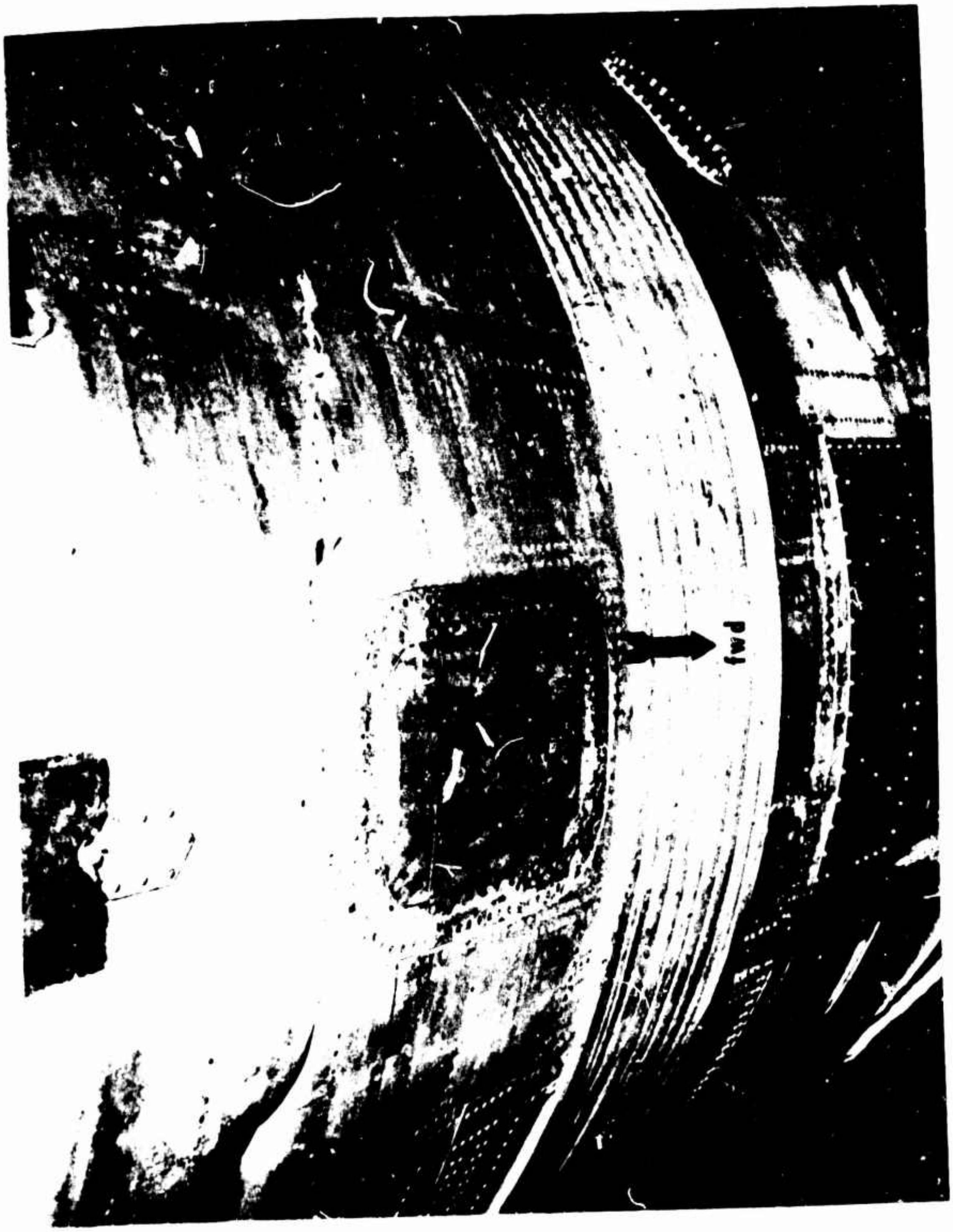

FIGURE 15 - Failure at Cycle No. 4556 - View of Lower Skin

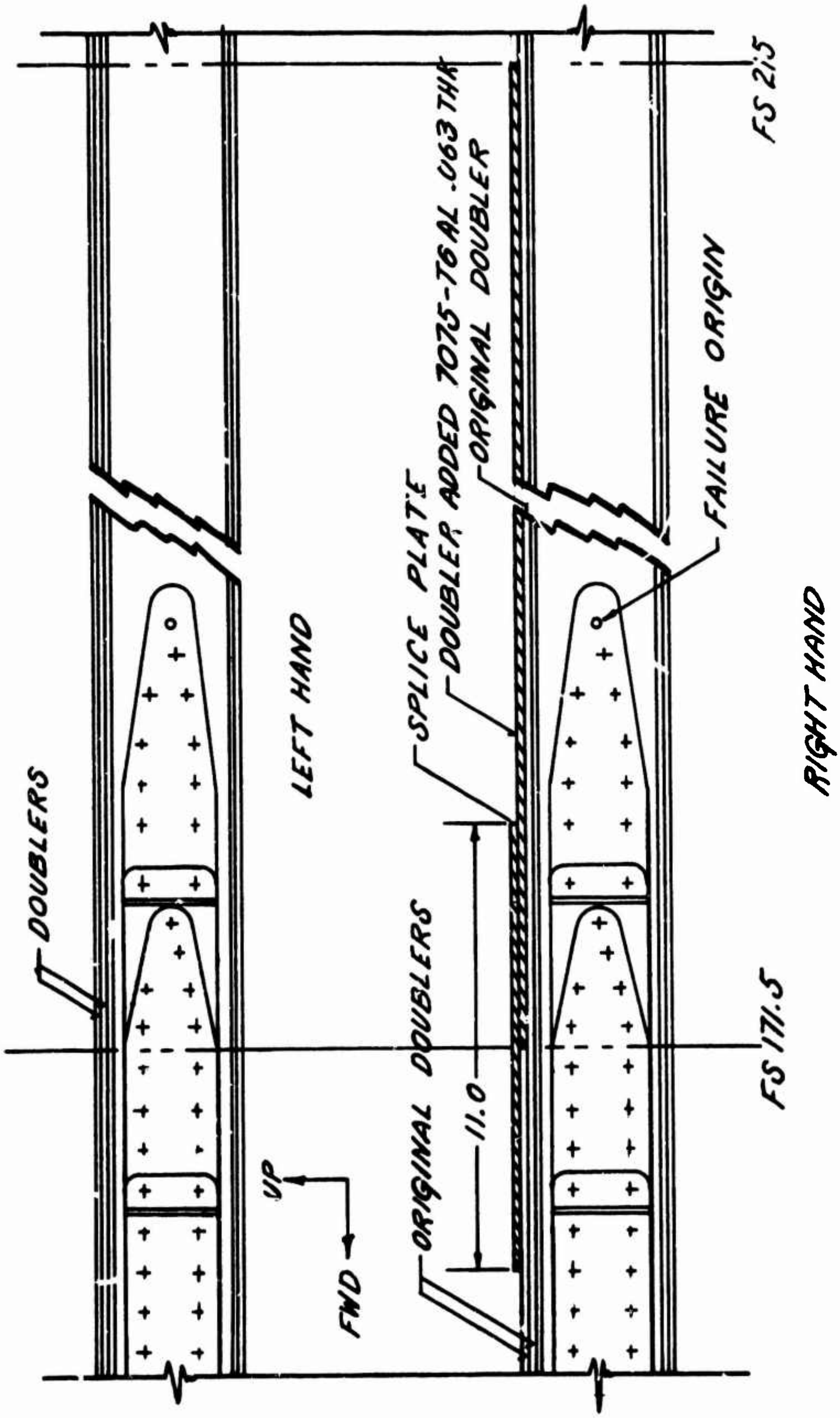

FIGURE 16 - R. H. Keel Beam With NADC Doubler Installed Compared to Unmodified L. H. Keel Beam

FIGURE 17 - Repair of Failure at Cycle No. 4556 - R. H. Skin

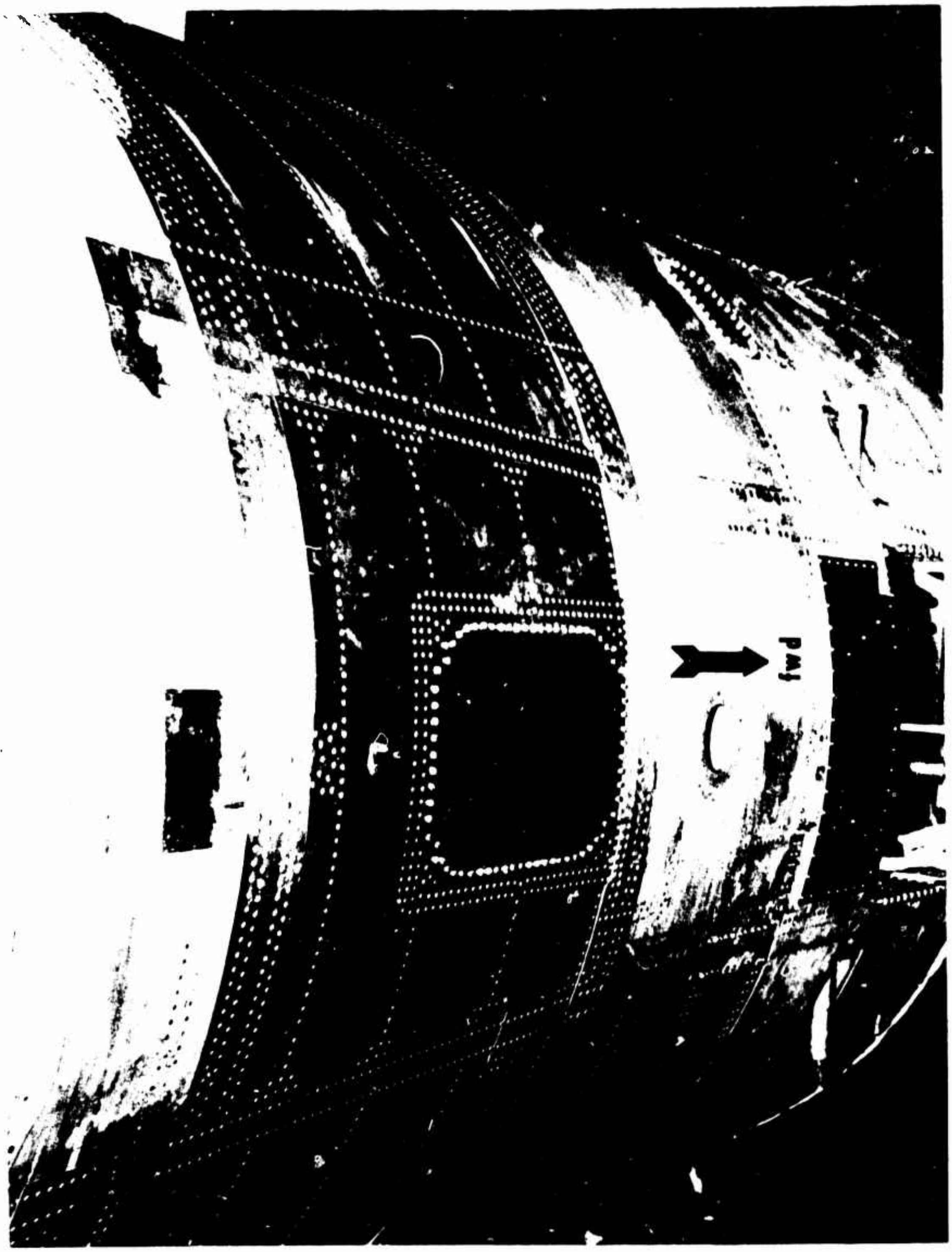

FIGURE 18 - Repair of Failure at Cycle No. 4556 - Bottom Skin

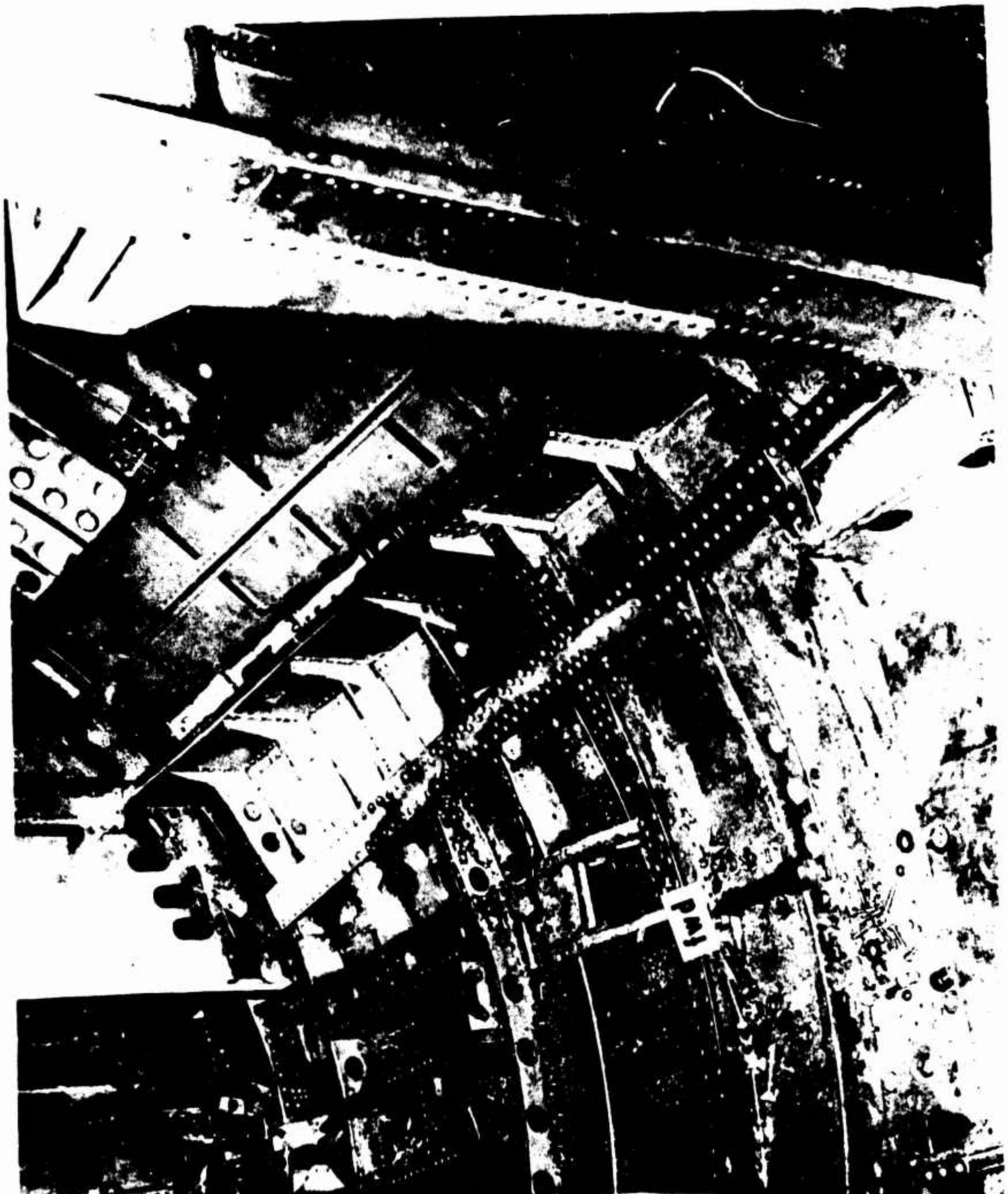

FIGURE 19 - Doubler Installed on R. H. Keel Beam After Failure at Cycle No. 4556

fuselage was stored outdoors and the crack became encrusted with corrosion products. Under the influence of the repeated loading in this program, the crack propagated until failure occurred during the 4556th cycle.

CONCLUSIONS

Based on the results of this test program, the E-2A/B fuselage is capable of sustaining the effects of 3000 arrested landings without structural modifications. The GAC fatigue analysis (reference (c)) indicates a life in excess of 4000 arrested landings.

The fuselage is capable of sustaining the effects of 2278 catapult launches without structural modifications and of sustaining the effects of 4000 catapult launches with a doubler installed on the flange of the right-hand keel beam.

RECOMMENDATIONS

It is recommended that the doubler shown in figure 16 be added to the right-hand keel beam of all E-2B airplanes after they have experienced 2200 catapult launches. This recommendation is conservative because it is based on the results of a test in which the test specimen was cracked prior to the start of testing. However, the crack was a fatigue crack which indicates that this area could be a fatigue sensitive area and the possibility exists that an uncracked test specimen may not have survived 8000 test cycles without failure. With the keel beam doubler installed, it is recommended that the catapult service limit for the fuselage be extended to 4000 catapult launches.

It is further recommended that the arrested landing service limit be extended to 3000 arrested landings. Consideration should be given to extending the service limit to 4000 arrested landings based on the results reported in reference (c).

REFERENCES

(a) GAC Report 3839.12, "Results of Catapulting Conditions Fatigue Tests," of 29 Mar 1965
(b) GAC Report 3840.12, "Results of Arrested Landing Fatigue Tests," of 7 Aug 1964
(c) GAC Report 3808.214, "Model E-2A Airplane Arresting Fatigue Study," of 22 Jun 1972

APPENDIX A

TEST DATA FOR CATAPULT AND ARRESTED LANDING FATIGUE TESTS OF THE MODEL E-2A/B AIRPLANE

LIST OF FIGURES

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LIST OF TABLES

S Y M B O L S

All symbols used in this appendix and in the text of the report are defined below:

FS . . . . fuselage station

FRL. . . . . fuselage reference line

LL . . . . limit load = 2/3 design ultimate load

Re . . . . resultant load

SIGN CONVENTION

The following sign convention is used: Distances and forces are positive when they are up, aft and to the left with respect to the reference axes. (See figure A-1.)

Positive bending moments produce compression in the top surface and left side of the fuselage.

Positive vertical shear results when the positive vertical loads are summed from a station of greater magnitude to one of lesser magnitude. Positive lateral shear results when the positive lateral loads are summed from a station of greater magnitude to one of lesser magnitude.

Positive torsion about the FRL results when a station of higher magnitude rotates clockwise in relation to a station of lower magnitude when viewed from aft.

REFERENCE AXES

X - axis: Lies in the plane of symmetry 100 inches below and parallel to the FRL.

Y - axis: Perpendicular to the plane of symmetry through the X - axis at FS 0.

Z - axis: Perpendicular to the X-Y plane through the intersection of the X and Y axes.

BASIC DATA

Arrested landing design gross weight - 40,660 pounds.

Catapulting design gross weight - 47,940 pounds.

Arrested landing test condition (reference (a)) - GAC condition 12F arrested landing, free-flight engagement with maximum hook load and mean run-out.

Catapulting test condition (reference (a)) - GAC condition 11C, catapult start of run II.

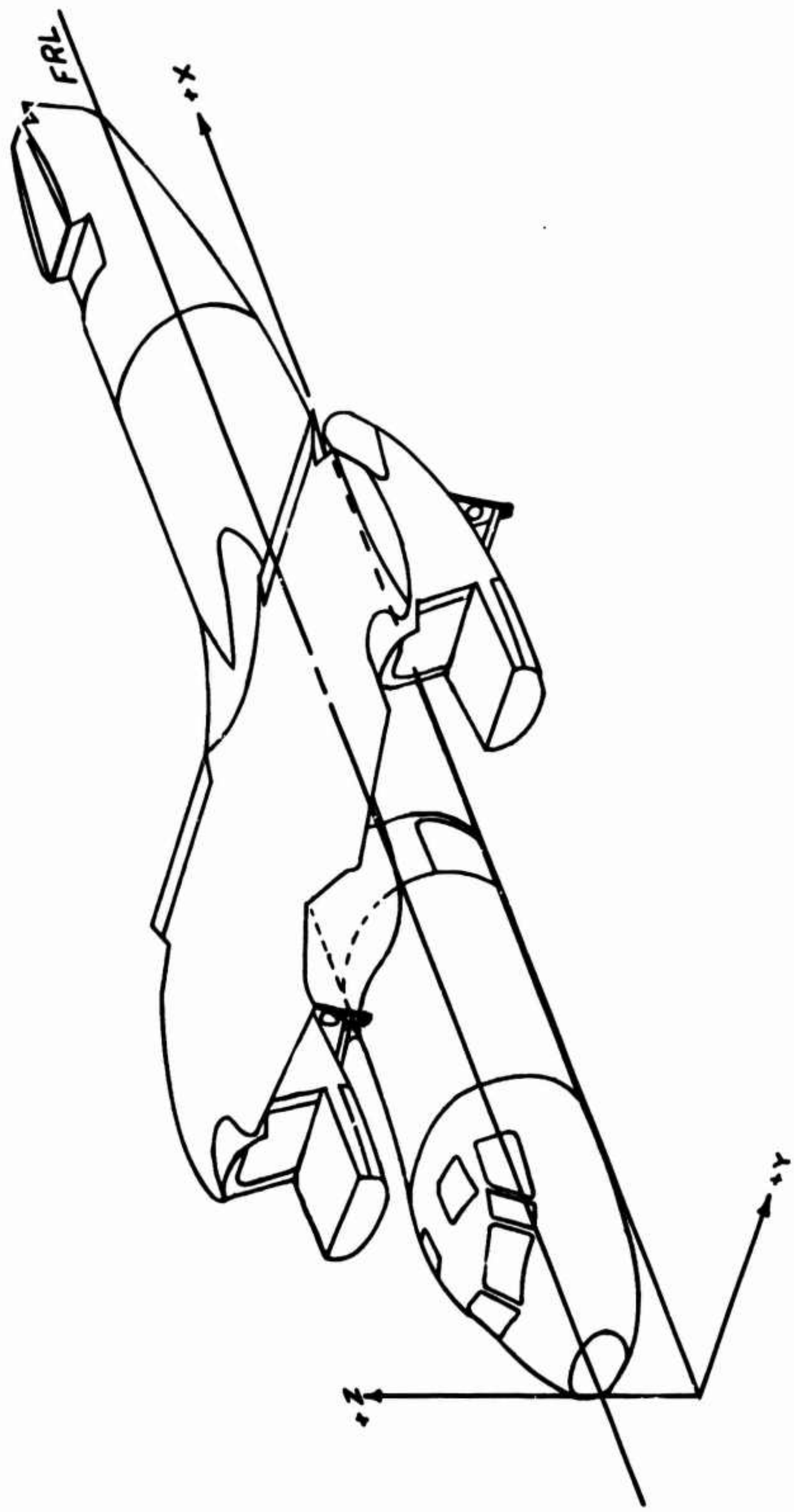


FIGURE A-1 - Reference Axes and Sign Convention

TEST LOADS

ARRESTED LANDING TEST

From page A-2 of referance (b), the arresting hook loads for condition 12F are as follows:

Limit Loads

X = +84,884
Y = ±39,800
Z = ∓2,455

The loads for each load level in the test spectrum are:

Load Level	% Limit A	% Limit S_R or S_L	Load (lbs) A	Load (lbs) S_R or S_L
1	68.3	37.7	58,000	±15,000
2	68.3	75.4	58,000	∓30,000
3	90.7	37.7	77,020	∓15,000
4	90.7	75.4	77,020	±30,000
5	117.8	37.4	100,030	±15,000
6	117.8	75.4	100,030	∓30,000

$A = \sqrt{X^2 + Z^2}$
$S_R = -Y$
$S_L = +Y$

The aft load A was applied at the resultant up angle of 1°39' with the FRL using a single actuator. A second actuator applied the sideloads.

The locations of all applied loads and reactions are shown on figure A-2. The magnitudes of all test loads are shown in table A-1. The distribution of loads is shown in tables A-2 and A-3. Comparisons of the design and test curves are shown in figures A-3 through A-8. These comparisons show good agreement in the critical area aft of the wing. It should be noted that the design curves show the effect of wing loads and moments on the fuselage by subdividing the loads and moments to the wing beams at FS 279 and FS 356 while the test curves indicate all wing loads and moments at the wing reference point FS 361.65. The two methods are, however, equivalent.

CATAPULT TEST

From pages A-8 and A-9 of reference (c), the nose gear loads for condition 11C are as follows:

Limit Tow Link Loads

X = -122,232 lbs
Y = ±2,500 lbs
Z = -46,807 lbs

Limit Nose Wheel Axle Loads

X = -168 lbs
Z = +10,066 lbs

A dummy nose gear was installed in the test specimen and the resultant of the tow link loads and the nose wheel axle loads were applied at a single point on the gear. The resultant of the X + Z components which is 127,795 pounds at a down angle of 16°71' with the FRL was applied with one actuator. The Y loads were applied with a second actuator.

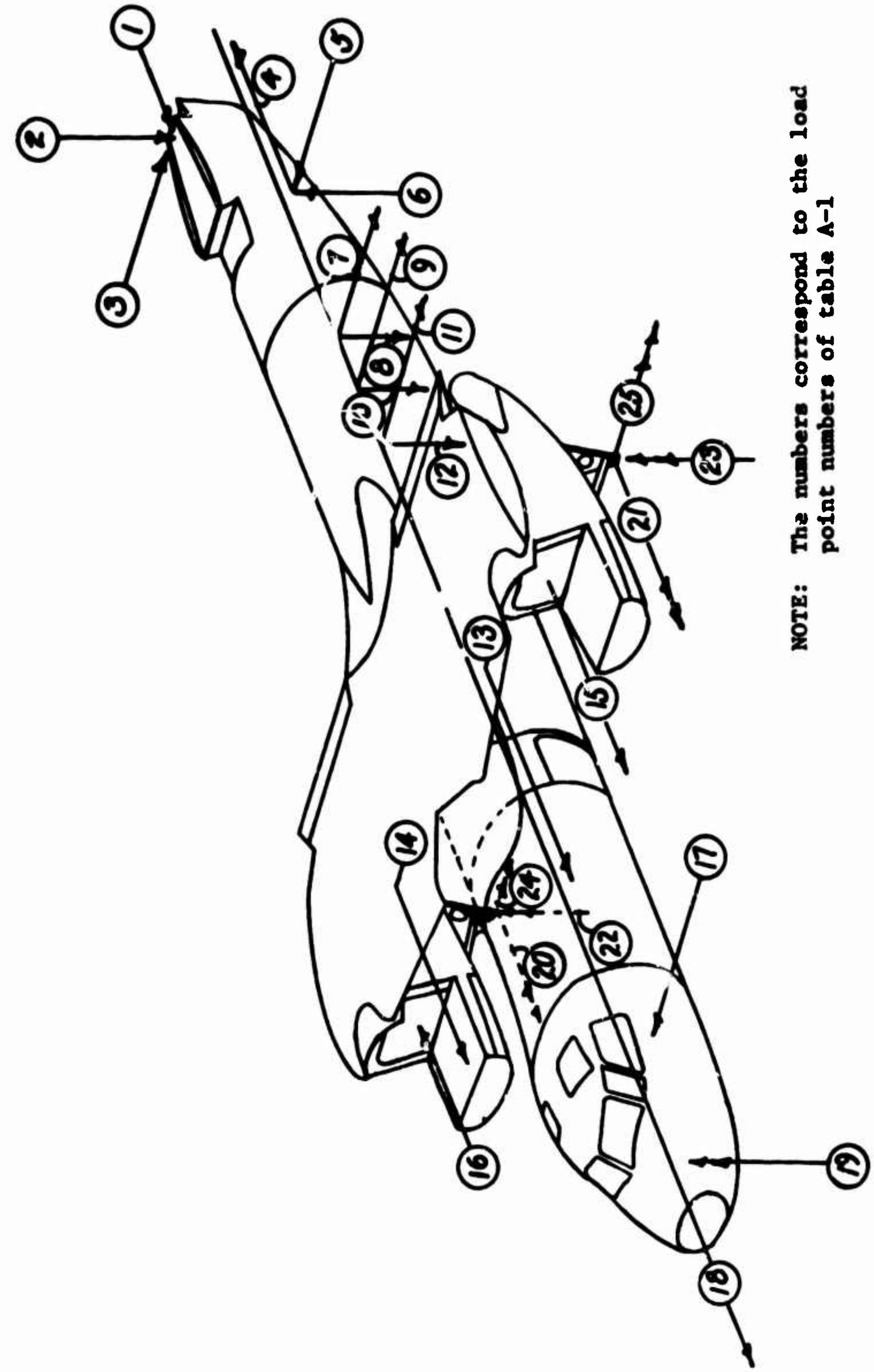
The locations of all applied loads and reactions are shown in figure A-9. The magnitudes of all test loads are listed in table A-4.

The distribution of loads is shown in tables A-5 and A-6. Comparisons of the design and test curves are shown in figures A-10 through A-15. These comparisons show good agreement in the critical area forward of the wing. It should be noted that since the design curves show the nose gear loads as a concentrated load at one point (page 14, reference (d)), the test curves also treat them as a concentrated load for the sake of clarity.

However, table A-5 also shows the load distribution at the actual points of entry into the fuselage, i.e., the shock strut trunnion at FS 64.50 and the drag brace trunnion at FS 126.50.

R E F E R E N C E S

(a) GAC Report No. 3803.3, "Ground Loading Conditions," of 15 Dec 1970
(b) GAC Report No. 3840.02, "Plan for Arrested Landing Conditions Fatigue Test," of 12 Jun 1962
(c) GAC Report No. 3839.02, "Plan for Catapulting Conditions Fatigue Test," of 13 Aug 1962
(d) GAC Report No. 3801.1, "Fuselage Net Loads," of 16 Jun 1967



NOTE: The numbers correspond to the load point numbers of table A-1

FIGURE A-2 - Load Points - Arrested Landing Test

TABLE A-1 - TEST LOADS - ARRESTED LANDING CONDITION - HOOK LOAD RIGHT

Load Point	Applied Loads Location, Inches X	Y	Z	Limit Load, Lbs X	Y	Z
1. Tail	655.44	0	143.63	-3200	0	0
2. Tail	655.44	0	143.63	0	0	-4800
3. Tail	655.44	0	143.63	0	+8000	0
4. Arresting Hook	663.22	0	89.78	+84884	0	0
5. Arresting Hook	663.22	0	89.78	0	-39800	0
6. Arresting Hook	663.22	0	89.78	0	0	+2455
7. Fuselage	555.00	0	100.00	0	+4675	0
8. Fuselage	555.00	0	100.00	0	0	-2250
9. Fuselage	499.50	0	120.00	0	+3900	0
10. Fuselage	499.50	0	100.00	0	0	-1000
11. Fuselage	432.50	0	90.00	0	+5610	0
12. Fuselage	432.50	0	100.00	0	0	-2000
13. Left Wing	361.65	+65.00	139.10	-19045	0	0
14. Right Wing	361.65	-65.00	139.10	-19045	0	0
15. Left Engine Mt.	361.65	+128.75	124.00	-27262	0	0
16. Right Engine Mt.	361.65	-128.75	124.00	+27262	0	0
17. Fuselage	141.00	0	76.61	0	-5000	0
18. Fuselage	141.00	0	100.00	-24684	0	0
Reaction Loads						
19. Nose Gear	64.50	0	81.25	0	0	+917
20. R. Main Gear	347.94	-126.76	60.44	-9455	0	0
21. L. Main Gear	347.94	+126.76	60.44	-9455	0	0
22. R. Main Gear	347.94	-126.76	60.44	0	0	+3399
23. L. Main Gear	347.94	+126.76	60.44	0	0	+3399
24. R. Main Gear	347.94	-126.76	60.44	0	+11307	0
25. L. Main Gear	347.94	+126.76	60.44	0	+11307	0

NOTES:

1. The X and Z loads applied to the arresting hook will be applied as the following resultant load at the following angle:

Re = 84,919#

θ = 1°39' up

2. For the hook load left condition the loads at load points 3, 5, 7, 9, 11, 15, 16, 17, 24 and 25 reverse direction.

TABLE A-2 - FUSELAGE VERTICAL SHEAR, BENDING MOMENT AND AXIAL LOAD - ARRESTED LANDING CONDITION 12 F-LIMIT LOAD

FS In	ΔX In	Vert Load Lbs	Vert Shear Lbs	Axial Load Lbs	ΔZ In	ΔBM Axial In-Lbs	ΔBM Vert In-Lbs	ΣBM In-Lbs
655.44		-4800		-3200	43.63	+139616		+139616
	53.69		-4800				-257712	
601.75		+2455		+84884	-12.00	+1018608		-118096
								+900112
	46.75		-2345				-109629	
555.00		-2250						+790883
	55.50		-4595				-255022	
499.50		-1000						+535861
	67.00		-5595				-374865	
432.50		-2000						+160996
	70.85		-7595				-538106	
361.65		6678		-57000		*+649684		-377110
								+272574
	220.65		- 917				-202336	
141.00				-26684	0.00			+70238
	76.50		- 917				-70151	
64.50		917						+87

*From Wing Loads

TABLE A-3 - FUSELAGE LATERAL SHEAR, BENDING MOMENT AND TORSION-ARRESTED LANDING CONDITION 12 F-LIMIT LOAD

FS In	ΔX In	Lat Load Lbs	Lat Shear Lbs	ΔBM In-Lbs	ΣBM In-Lbs	ΔZ In	ΔTorsion In-Lbs	ΣTorsion In-Lbs
655.44		8000				43.63	-349040	-349040
	53.69		8000	429520	*429520			
601.75		-39800		-2446506	-2016986	-10.22	-406756	-755796
	46.75		-31800	-1486650				
555.00		4675			-3503636			
	55.50		-27125	-1505437				
499.50		3900			-5009073	20.00	-78000	-833796
	67.00		-23225	-1556075				
432.50		5610			-6565148	-10.00	56100	-777696
	70.85		-17615	-1248023 *6709921				
361.65		22615			-7813171 -1103250	-39.56	+894649	+116953
	220.65		+5000	+1103250				
141.00		-5000			0	-23.39	-116953	0

*From Wing Loads

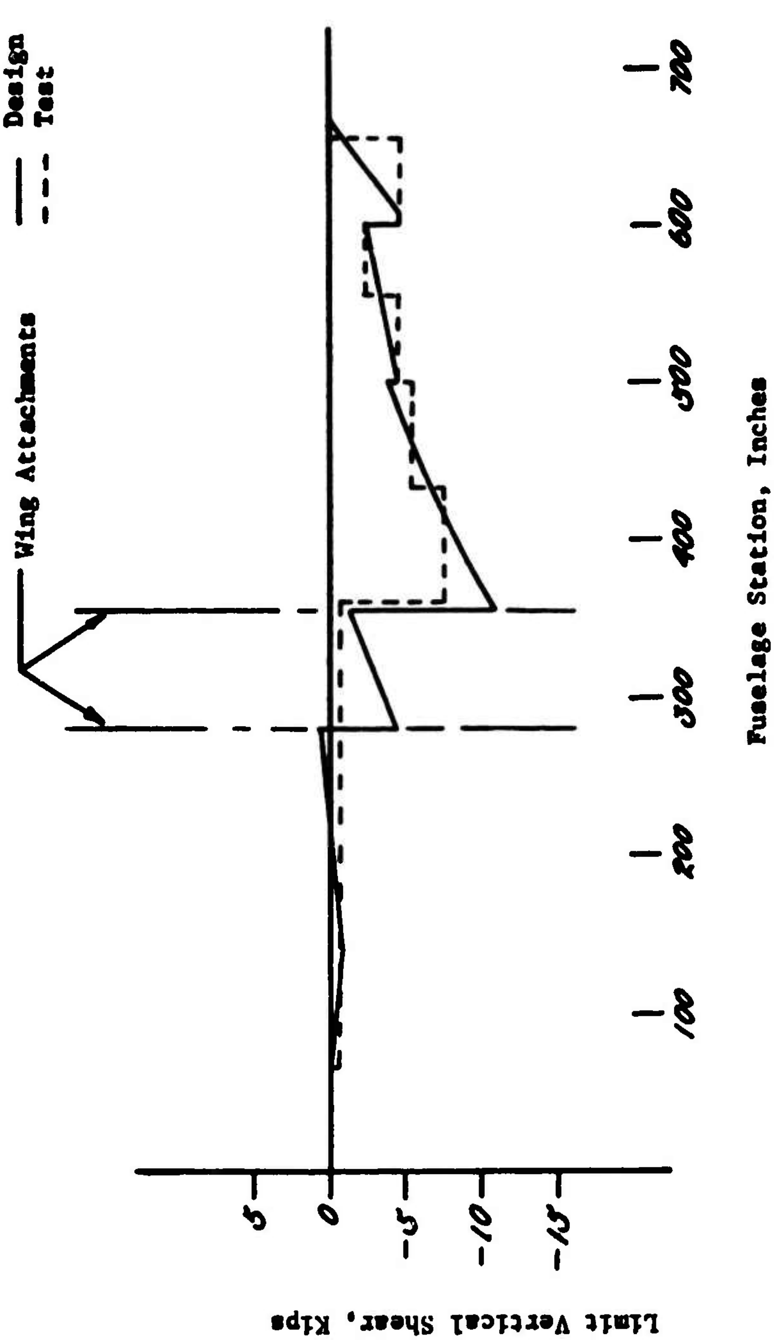


FIGURE A-3 - Fuselage Vertical Shear Distribution - Arrested Landing Condition

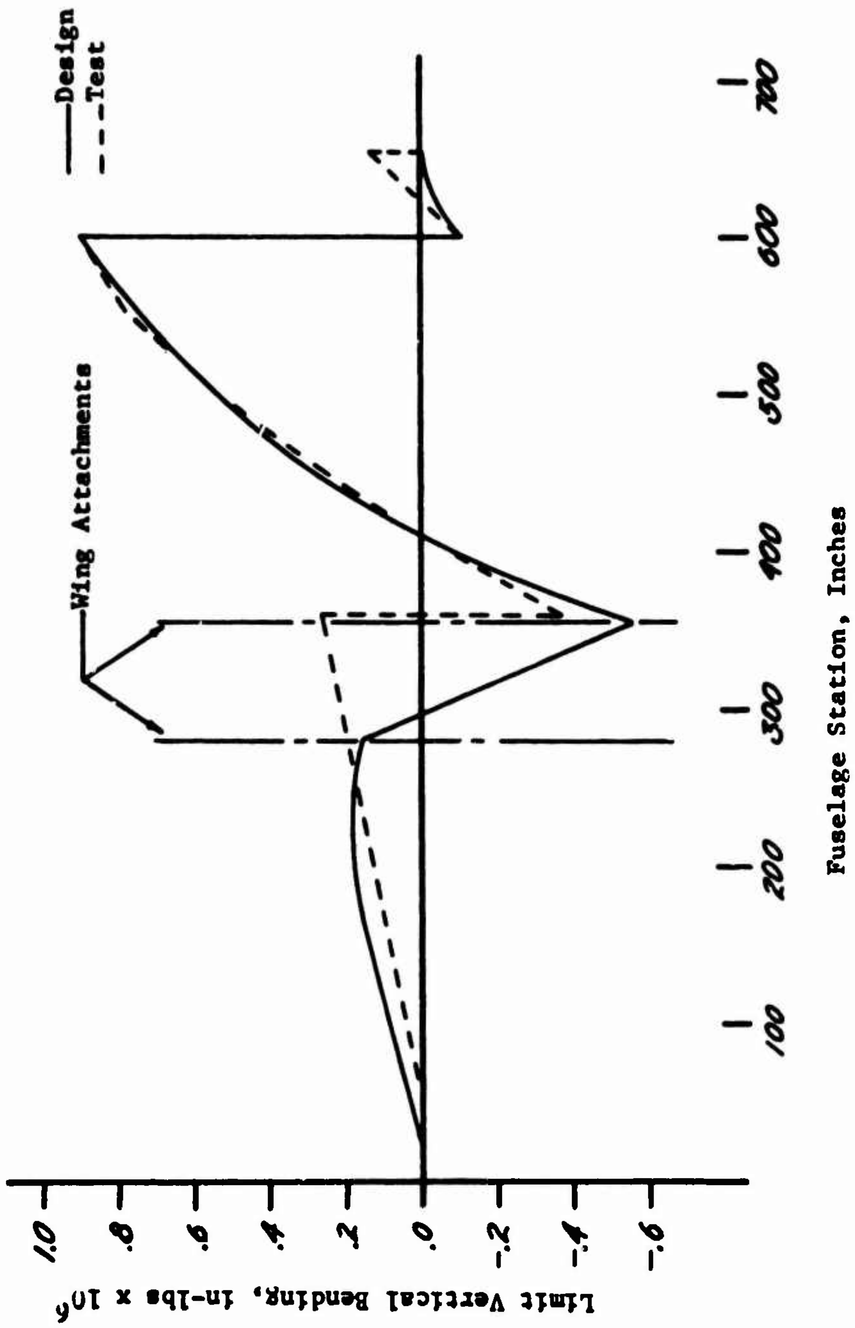


FIGURE A-4 - Fuselage Vertical Bending Moment Distribution - Arrested Landing Condition

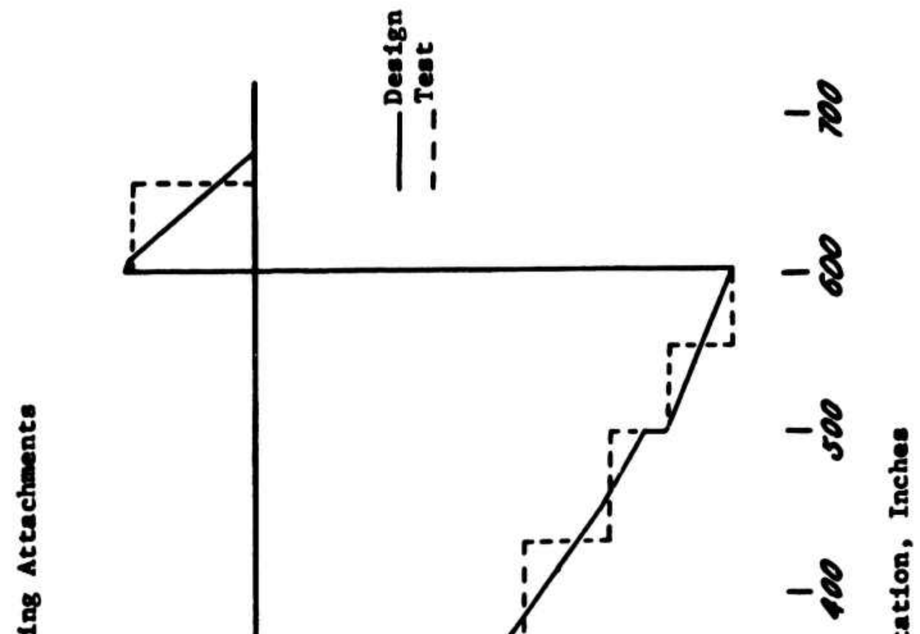


FIGURE A-5 - Fuselage Lateral Shear Distribution - Arrested Landing Condition

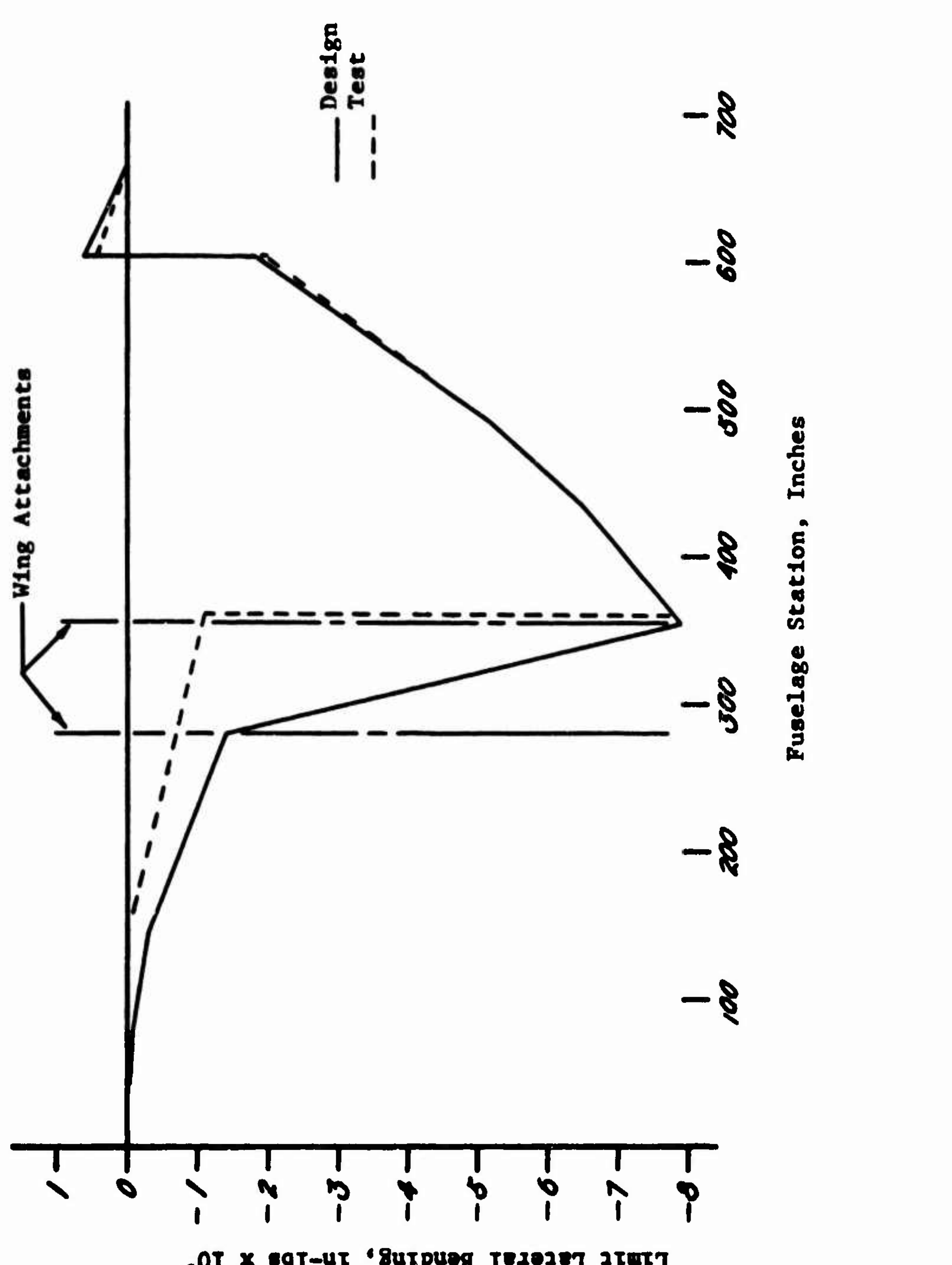


FIGURE A-6 - Fuselage Lateral Bending Moment Distribution - Arrested Landing Condition

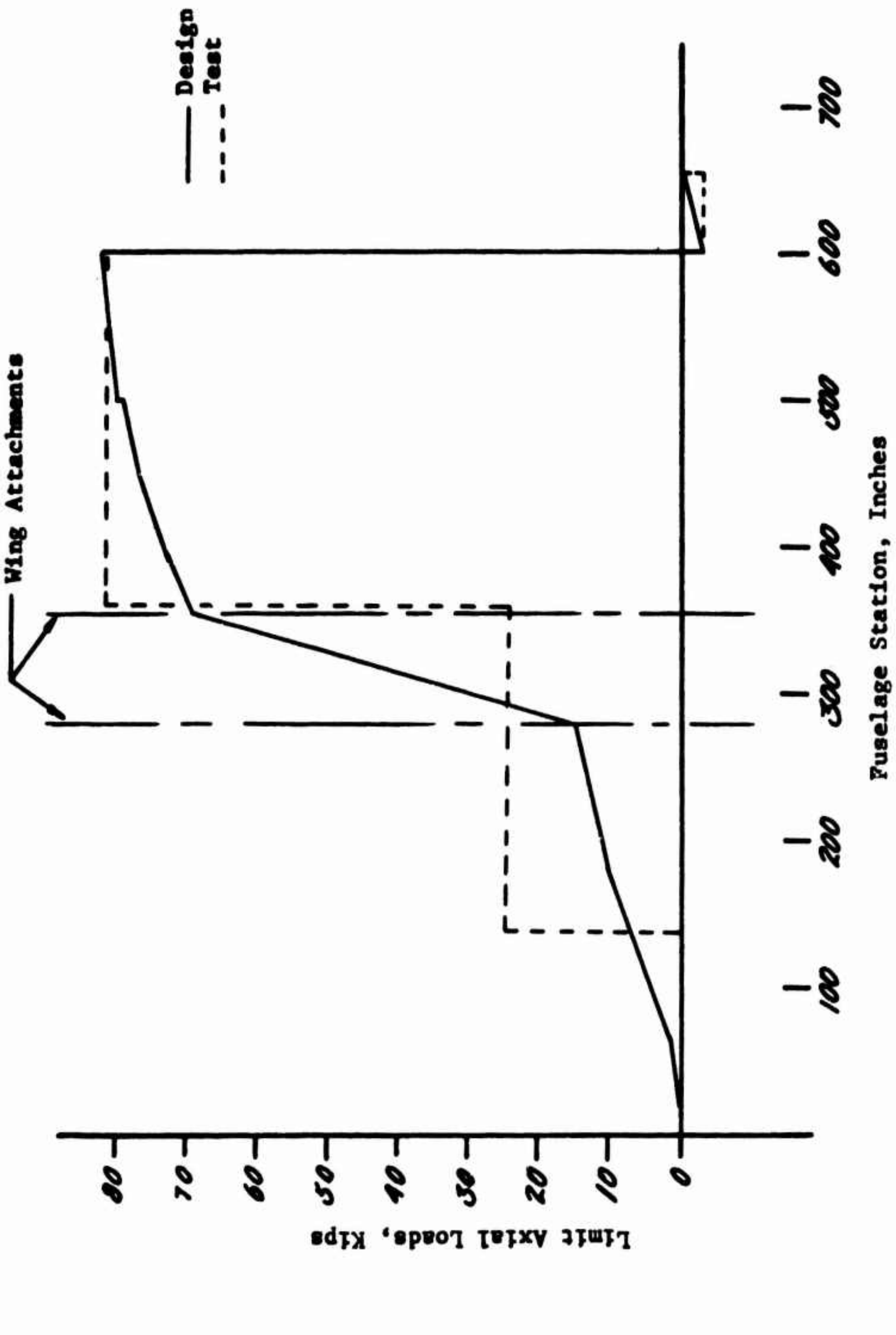


FIGURE A-7 - Fuselage Axial Load Distribution - Arrested Landing Condition

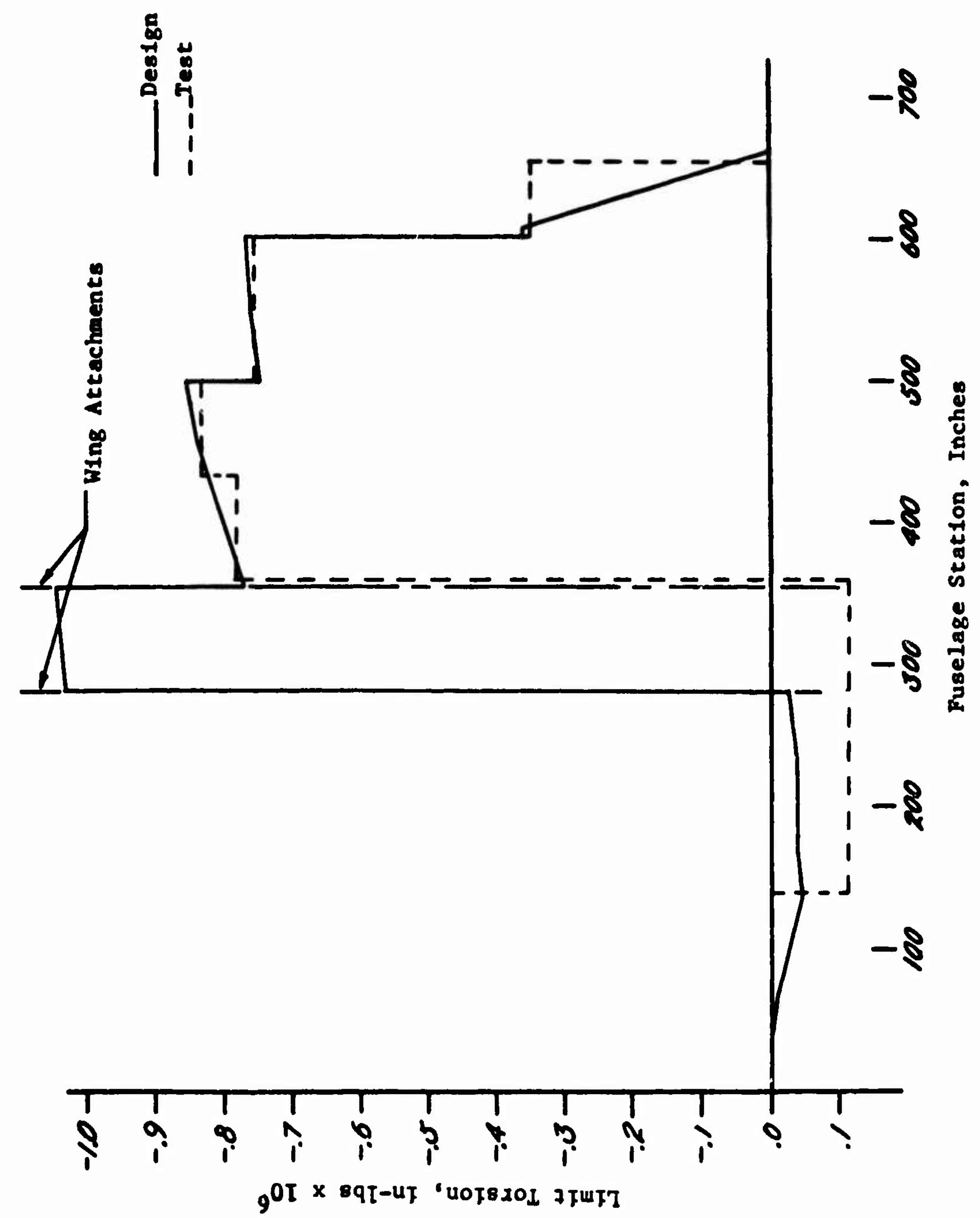
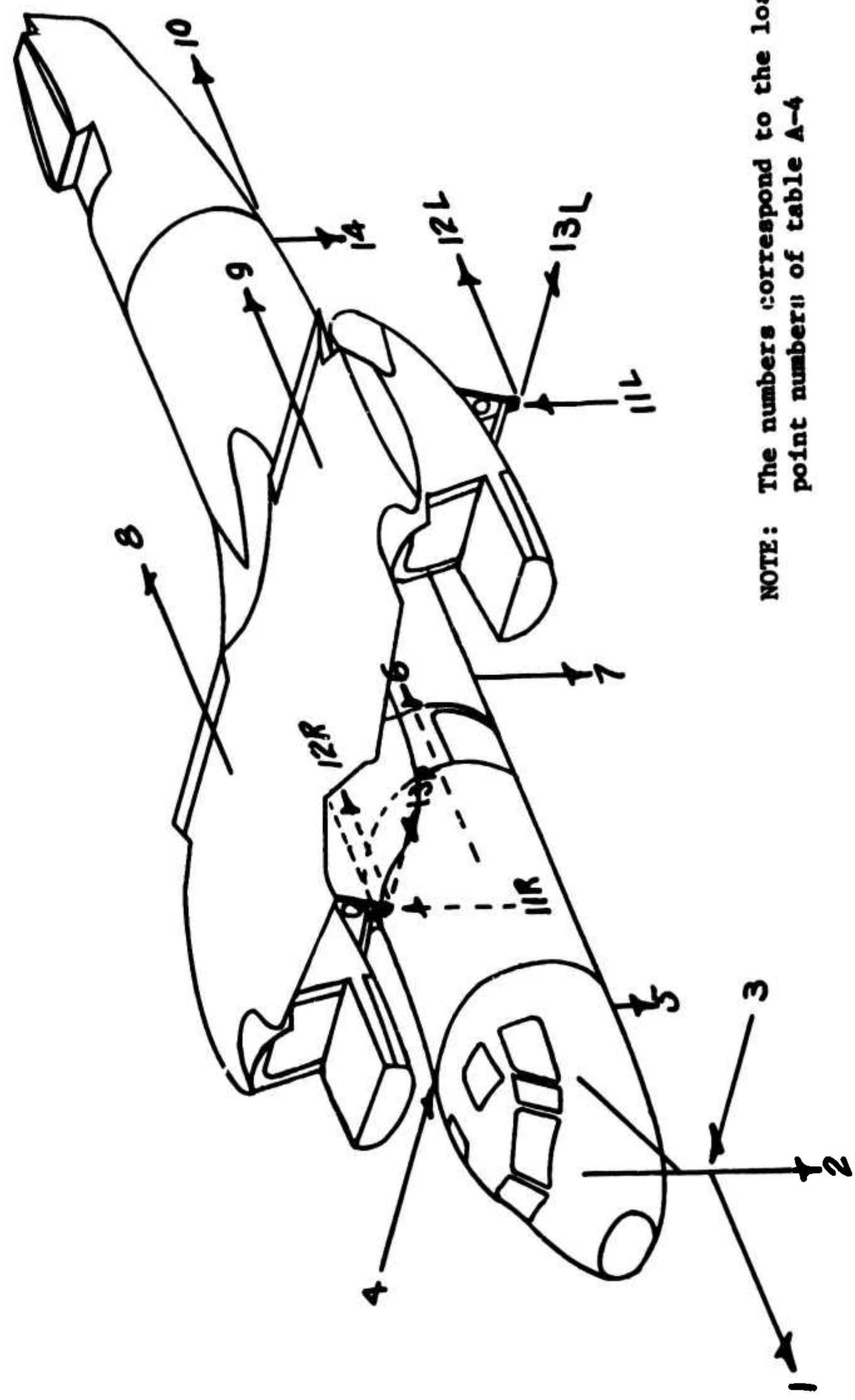


FIGURE A-8 - Fuselage Torsional Moment Distribution - Arrested Landing Condition



NOTE: The numbers correspond to the load point numbers of table A-4

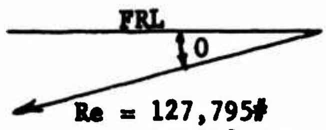
FIGURE A-9 - Load Points - Catapult Test

TABLE A-4 - TEST LOADS - CATAPULT CONDITION 11c - SIDE LOAD RIGHT

Load Point	Applied Load					
	Location, Inches			Limit Load, Lbs		
	X	Y	Z	X	Y	Z
1. Dummy Nose Gear	58.37	0	31.98	-122400	0	0
2. Dummy Nose Gear	58.37	0	31.98	0	0	-36741
3. Dummy Nose Gear	58.37	0	31.98	0	-2500	0
4. Fuselage	141.00	0	148.00	0	+1107	0
5. Fuselage	141.00	0	100.00	0	0	-3400
6. Fuselage	203.00	0	95.28	+15000	0	0
7. Fuselage	356.00	0	100.00	0	0	-6500
8. Right Wing	347.94	-65.00	142.70	+26281	0	0
9. Left Wing	347.94	65.00	142.70	+26281	0	0
10. Arresting Hook	601.75	0	88.00	+34000	0	0
Reaction Loads						
11L. L. Main Gear	347.94	126.76	60.44	0	0	+29086
11R. R. Main Gear	347.94	-126.76	60.44	0	0	+27326
12L. L. Main Gear	347.94	126.76	60.44	+12310	0	0
12R. R. Main Gear	347.94	-126.76	60.44	+8528	0	0
13L. L. Main Gear	347.94	126.76	60.44	0	+696	0
13R. R. Main Gear	347.94	-126.76	60.44	0	+696	0
14. Fuselage	555.00	0	100.00	0	0	-9771

Notes:

1. The X and Z loads applied to the dummy nose gear will be applied as the following resultant load:



Re = 127,795#
0 = 16.71° down

This load is the resultant of the catapult tow link loads and wheel axle loads.

2. When the nose gear side load is applied to the left the loads at load points 3,4,13L and 13R reverse direction and the loads at points 11L and 12L are interchanged with those at points 11R and 12R.

TABLE A-5 - FUSELAGE VERTICAL SHEAR, BENDING MOMENT AND AXIAL LOAD

CATAPULT CONDITION 11c - LIMIT LOAD

F S in.	Δ X in.	Vert Load Lbs	Vert Shear Lbs	Axial Load Lbs	Δ Z in.	Δ BM Axial in - lbs	Δ BM Vert in - lbs	Σ BM in - lbs
601.75				+34000	-12.00	+408000		+408000
555.00		-9771						+408000
	199.00		-9771				-1944429	
356.00		-6500						-1536429
	8.06		-16271				-131144	
347.94		+56412						-1667573
				+73401	+19.35	-1420309		-3087882
	144.94		+40141				+5818036	
203.00				+15000	-4.72	+70800		+2730154
	62.00		+40141				+2488742	+2800954
141.00		-3400						+5289696
	14.50		+36741				+532744	
126.50		-56728		-143036	-34.75	-4970501		+5822440
	62.00		-19987				-1239194	+851939
64.50		+19987		+20636	-18.75	+386925		-387255
								-330

Note: If the nose gear loads are treated as a concentrated load (page 14 reference (e)) the distribution of loads forward of FS 141 is as follows:

F S in.	Δ X in.	Vert Load Lbs	Vert Shear Lbs	Axial Load Lbs	Δ Z in.	Δ BM Axial in - lbs	Δ BM Vert in - lbs	Σ BM in - lbs
141.00		-3400						+5289696
	72.20		+36741		-64.89		+2652700	
68.80		-36741		-122400		-7942396		+7942396
								0

TABLE A-6 - FUSELAGE LATERAL SHEAR, BENDING MOMENT AND TORSION

CATAPULT CONDITION 11c - LIMIT LOAD

F S in.	ΔX in.	Lat Load Lbs	Lat Shear Lbs	ΔBM in - lbs	ΣBM in - lbs	ΔZ in.	ΔTorsion in - lbs	ΣTorsion in - lbs
347.94		+1393			-479517 (1)			+223186 (2)
	206.94		+1393	+288267				
141.00		+1107			-191250	+48.00	-53136	+170050
	76.50		+2500	+191250				
64.50		-2500			0	-68.02	-170050	0

(1) From MLG X loads

(2) From MLG Z loads

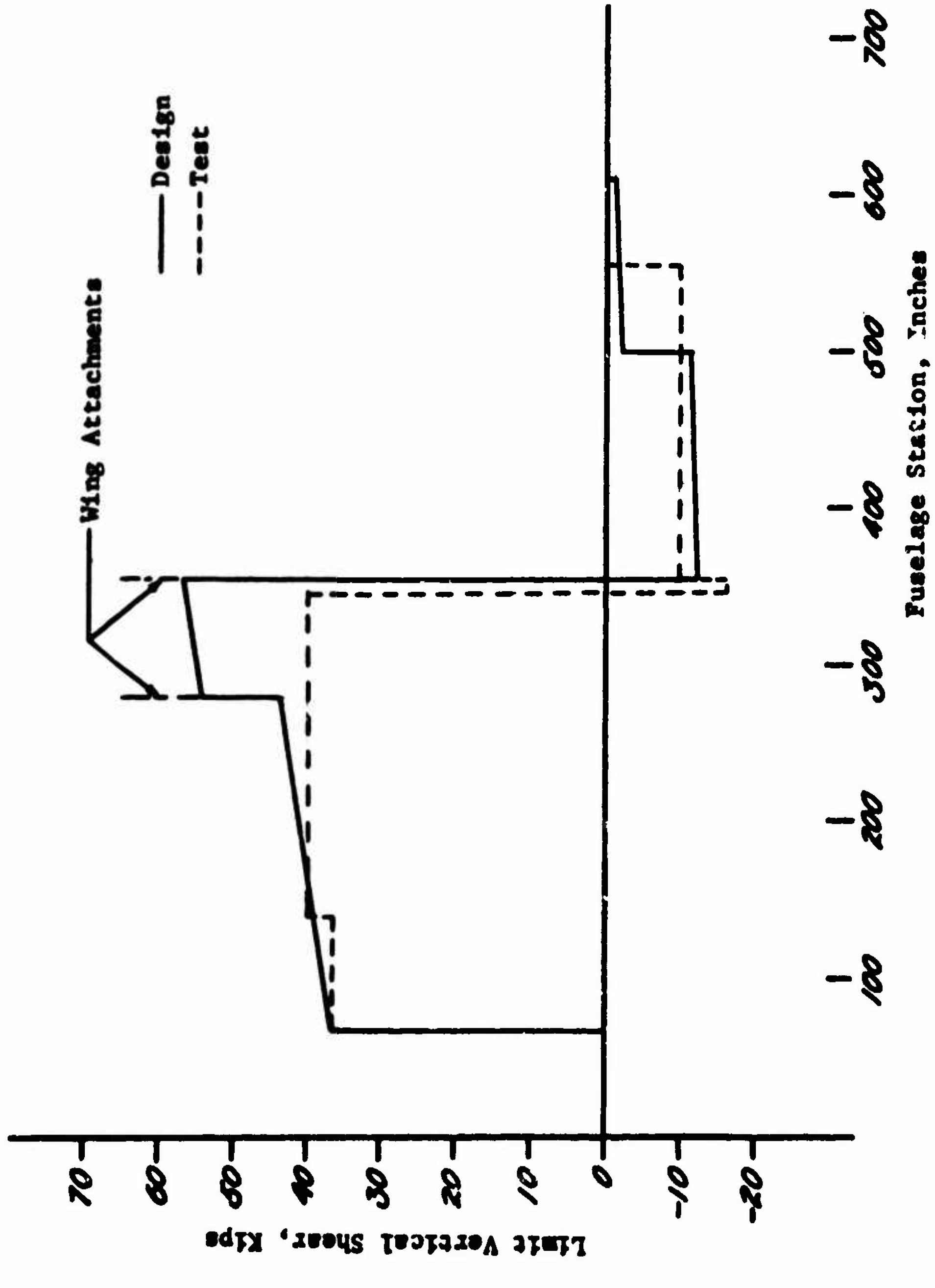


FIGURE A-10 - Fuselage Vertical Shear Distribution - Catapult Condition

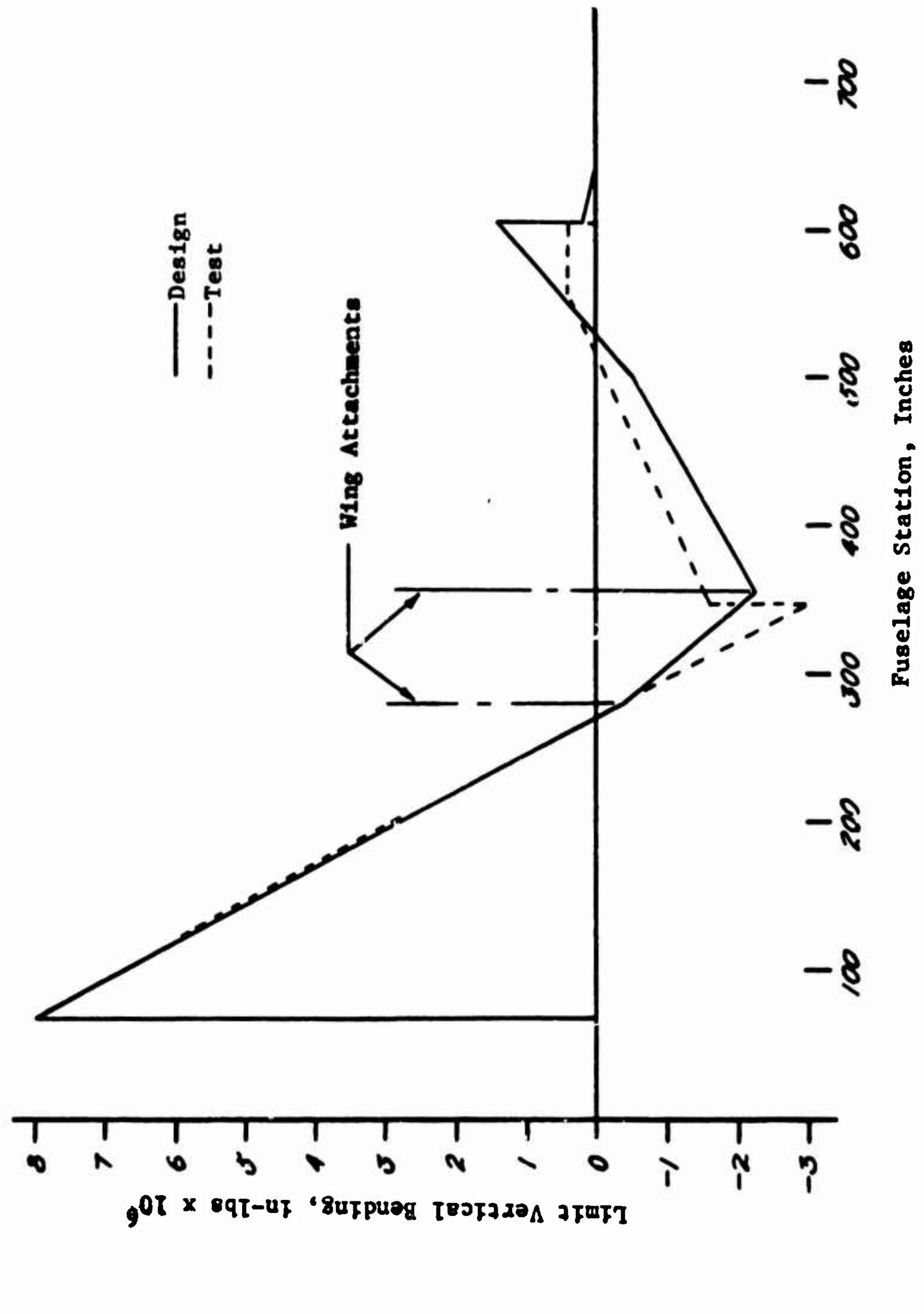


FIGURE A-11 - Fuselage Vertical Bending Moment Distribution - Catapult Condition

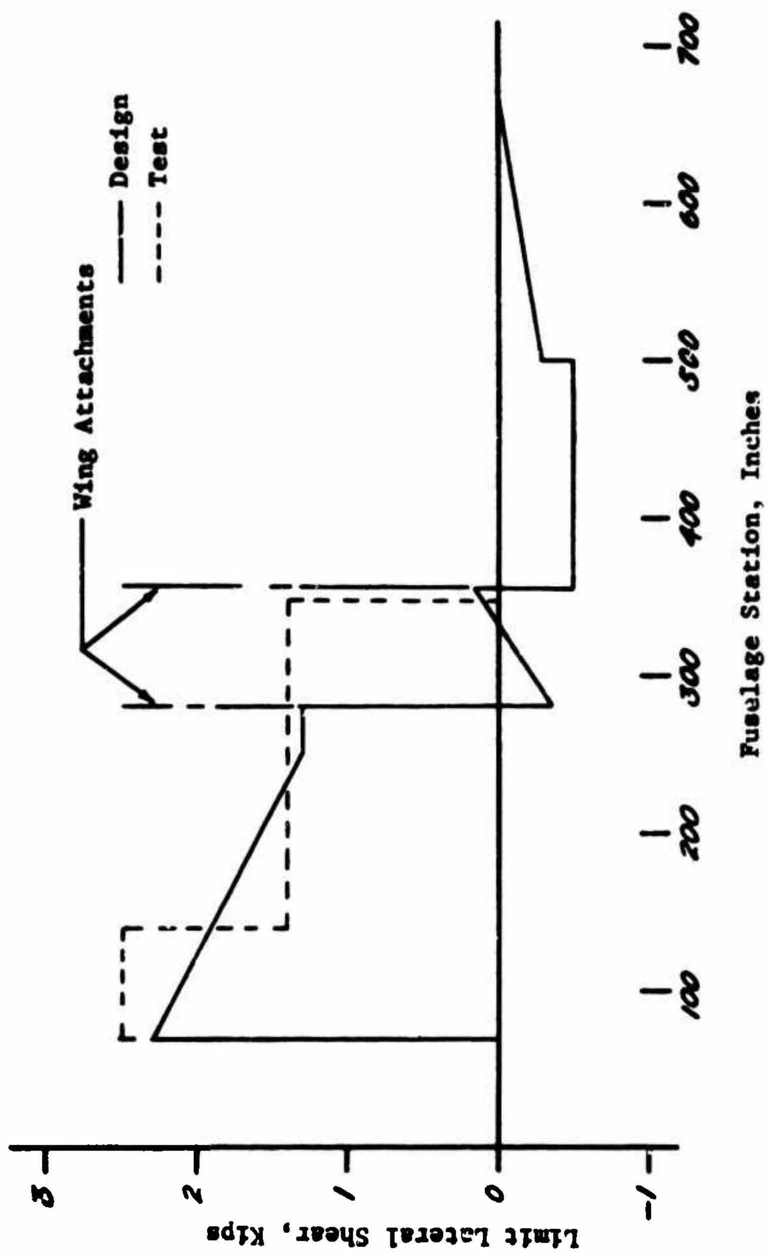


FIGURE A-12 - Fuselage Lateral Shear Distribution - Catapult Condition

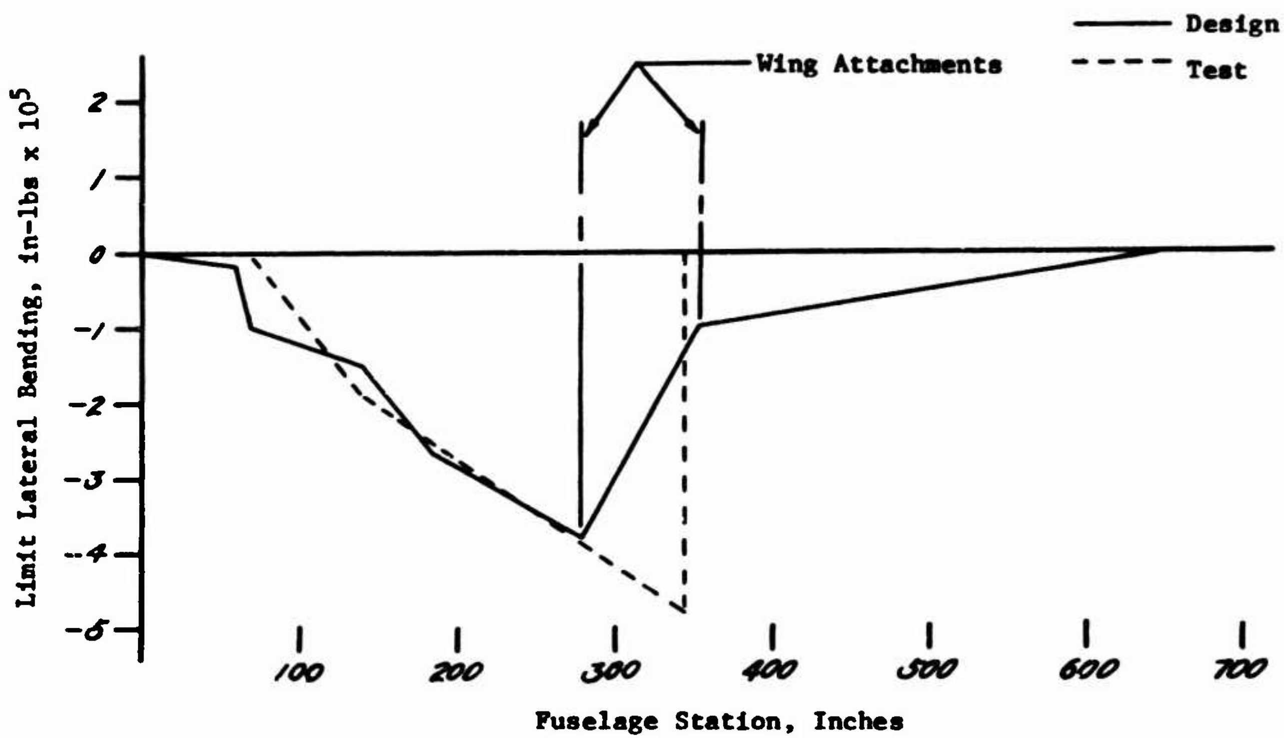


FIGURE A-13 - Fuselage Lateral Bending Moment Distribution - Catapult Condition

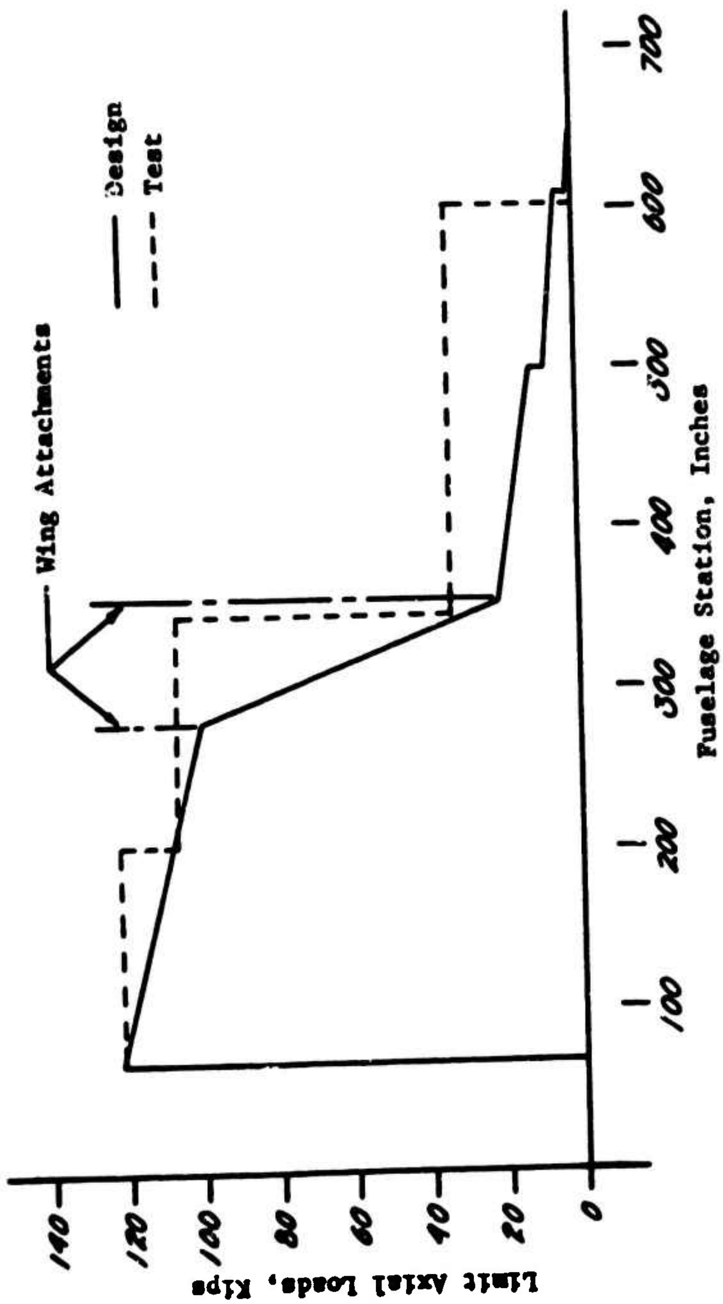


FIGURE A-14 - Fuselage Axial Load Distribution - Catapult Condition

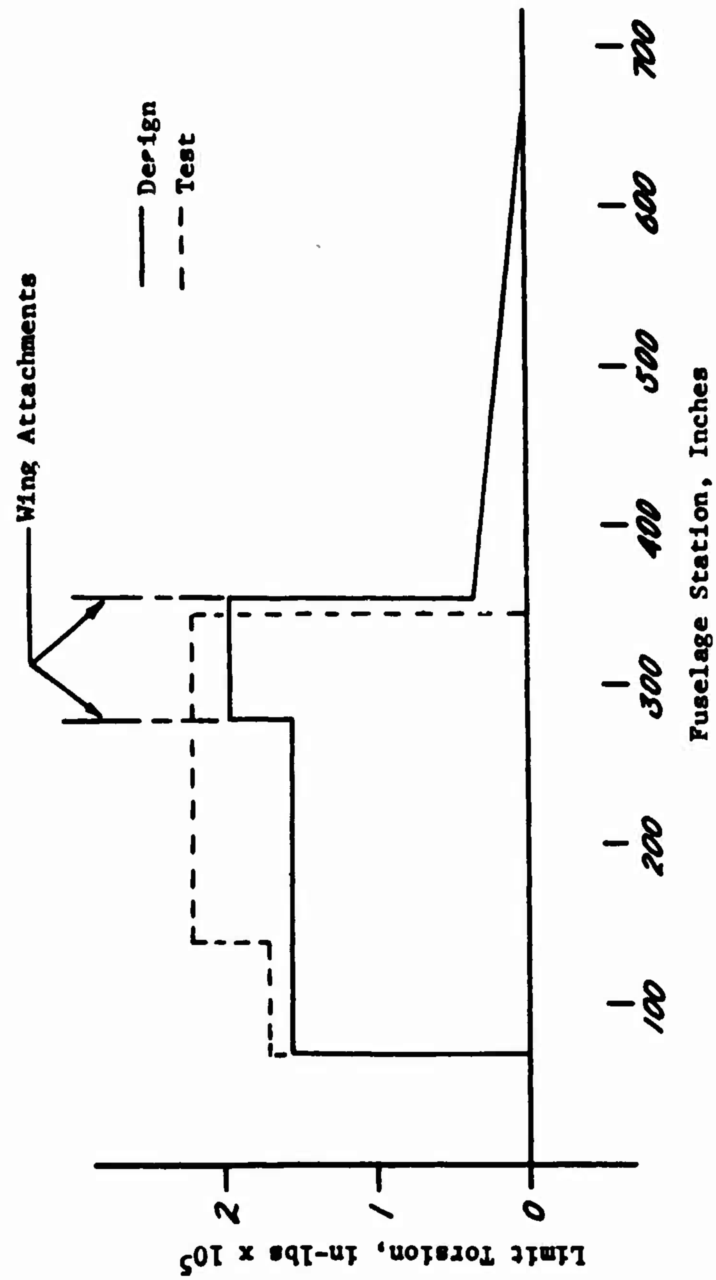


FIGURE A-15 - Fuselage Torsional Moment Distribution - Catapult Condition